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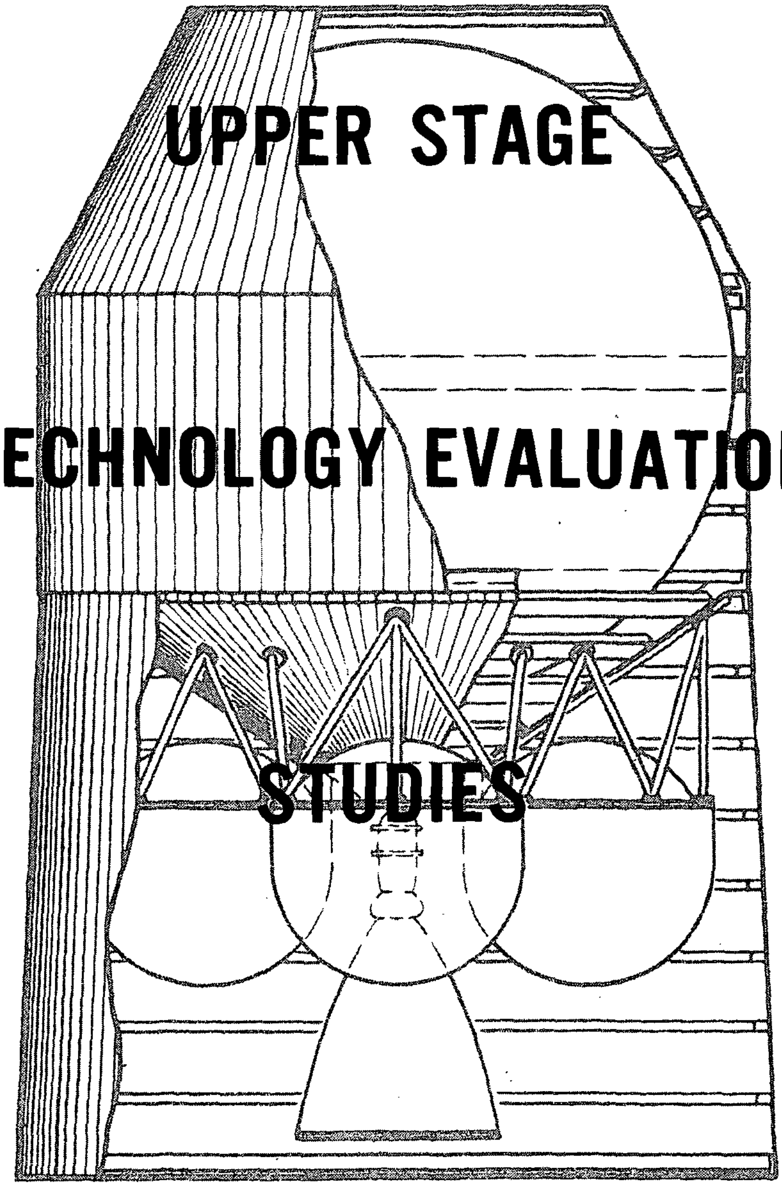
UPPER STAGE TECHNOLOGY EVALUATION STUDIES



AUGUST 31, 1972

SPACE DIVISION CHRYSLER CORPORATION

TECHNICAL REPORT TR-AP-72-6

UPPER STAGE TECHNOLOGY EVALUATION STUDIES

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Prepared for

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Washington, D. C., under Contract NASW-2294

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FOREWORD

This report was prepared by the Chrysler Corporation Space Division, New Orleans, Louisiana, and contains the results of a study performed for the National Aeronautics and Space Administration, Office of Advanced Research and Technology, under contract NASW-2294, "Upper Stage Technology Evaluation Studies".

ACKNOWLEDGEMENTS

Upper Stage Technology, Evaluation Studies were performed under the direction of William Cohen, NASA Headquarters, telephone (202) 755-2400.

Propulsion company support was provided by Rocketdyne Division, North American Rockwell Corporation. Specific contributions included parametric performance data for ultra-high chamber pressure oxygen-hydrogen engines. This support was valuable to the study and was much appreciated.

Chrysler Corporation Space Division personnel responsible for major contributions to the study included:

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L. V. Cressy - Cost Model Development

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Section 1

INTRODUCTION

In the performance of this contract, Chrysler conducted various studies to evaluate advanced technology relative to chemical upper stages and orbit-to-orbit stages. The objectives of these studies were essentially three-fold: 1) to provide NASA with data to provide a quantitative basis for making decisions for future allocation of available resources in the continuing development of propulsion technology, 2) parametric data for use in other studies, and 3) quantitative information to be used in the selection and direction of other studies.

To meet the objectives of this contract, three different tasks were undertaken. Two of these tasks (1 and 3) were small in nature, and dealt entirely with the modification and documentation of the stage sizing computer program developed for a use in contract NAS7-790(1), while the third and main task (Task 2) of this contract dealt primarily with the application of the computer program in accomplishing various task assignments made by NASA. Six work orders (summarized in table 1-1) were assigned by NASA under Task 2. Analytical studies were performed under each of these assignments, with the exception of work orders 3 and 4, which called for additional computer program modifications.

Under Task 1, a cost model, developed specifically for preliminary design tradeoff studies and long-range program cost projections, was included in the stage sizing computer program. This permitted parametric sensitivity and optimization analyses to be performed on the basis of cost as well as weight during Task 2.

Although the majority of the effort expended under this contract was associated with the development of analytical data under Task 2, Work Orders 3 and 4 involved the modification of the computer program. The objectives of these two assignments were to provide the capability to handle other types of potential propellants. Under these work orders the thermal routines in the sizing program were modified to permit the use of storable propellants, and 14 new monopropellant stage geometries were added to the sizing program.

Documentation of the Tasks and Work Orders, which were concerned with the sizing program itself, has been included in the program utilization report under Task 3, and published as two separate reports (2, 3).

Table 1-1. Task 2 Work Orders

Work Order	Description of Work
1	Developed LH_2/LOX stage data
2	Developed data to indicate stage sensitivity to engine tolerances
3	Modified thermal routines to accomodate storable propellants
4	Added stage geometries to computer program for mono-propellant configurations
5	Determined the relative gain obtainable through improvement of stage mass fraction
6	Studied future propulsion concepts.

The work conducted under the remaining four work orders (1, 2, 5 and 6) issued for Task 2, quantitatively assessed the merits of advances in various technologies related to improved performance of space vehicles. The results of the analyses conducted under these work orders are contained in this report.

The purpose of Work Order 1 (of Task 2) was to assess the merits of technology advances in space propulsion in the areas of increased chamber pressures and reduced engine weight. This was accomplished by developing a set of general data which covered a wide range of engine specific impulses and weights, and stage sizes. In addition, more detailed analyses were undertaken to determine the implications of ultra-high chamber pressures (up to 10,000 psi), moderate increases in chamber pressures (up to 3,000 psi) and the reduction of engine weight.

At the completion of Work Order 1, it was mutually decided that it would be advantageous to investigate other methods to improve the mass fraction of a stage. Such investigation provided a comparison of the various means of increasing stage performance. Therefore, Work Order 2, "Stage Sensitivity to Engine Tolerances" was replaced with two additional work orders (5 and 6).

Under Work Order 5, "Relative Gains Obtainable Through Improvement of Mass Fraction," a set of quantitative data was developed to show the relative gains which might be realized through the use of more advanced structural material, a reduction in subsystem weights, etc.

The purpose of the last work order (6), was to determine the relative merits of various distant future propulsion concepts, such as the use of metallic and atomic hydrogen, activated oxygen and compounds of activated helium, and metastable activated helium. However, because thermochemical property data for these propellants were difficult to obtain, it was possible to investigate only two of these propellants: metallic hydrogen and activated helium.

The results of the analyses of these future propellants, as well as the results of the other analytical work conducted under Task 2, are presented in the remainder of this report.

Section 2

TASK 2 - WORK ORDER 1., LOX/HYDROGEN STAGE DATA

2.1 GENERAL

The objective of this work order was to provide quantitative data which could be used in assessing the merits of technology advances in space propulsion, such as increases in chamber pressure, or decreases in engine weight. This was accomplished by the development of general data for a range of specific impulses and engine weights and several different missions. In addition, more detailed data were developed for ultra-high chamber-pressure increases, moderate chamber-pressure increases, and reductions in engine weights. These data are discussed in sections 2.2 through 2.5.

2.2 GENERAL LOX/HYDROGEN STAGE DATA

A set of generalized parametric data was developed for LOX/Hydrogen stages which relates engine specific impulse and weight to stage size and cost. These data are presented for a wide range of specific impulses and engine weights, and for three missions having different constraints. The three missions considered were: 1) earth orbit to synchronous orbit and return; 2) a single-burn Mars planetary orbit insertion; and 3) a two-burn Mars planetary mission.

Data presented in these generalized curves can be used to evaluate any technology advancement which will result in an improvement in either engine weight or specific impulse. In addition to being useful in estimating stage weight and cost, the curves can be utilized to determine partial derivatives (i.e., $\frac{W_{stage}}{Isp}$, $\frac{Cost}{W_{eng}}$, etc.) and the break-even point for certain propulsion technology developments. An example would be to determine how much money could be saved in the area of program costs if the engine weight was decreased by 10 percent through the use of lighter turbomachinery. Other examples of the use of these curves are described in section 2.2.4, Sample Solutions.

The generalized data developed in this task are presented in subsections 2.2.1 through 2.2.4.

2.2.1 Data and Assumptions

Throughout the study, certain constraints, guidelines and pertinent design data were used. These are summarized for the generalized data in this section. Table 2-1 gives the design constraints used for both the synchronous and Mars missions. Table 2-2 presents the prime structure data used in computing the weights of the shell and thrust structure. Table 2-3 summarizes the assumed tankage design data, including pertinent thermal and meteoroid protection data. These data are shown for each mission. The weights assumed for the astrionics systems and for other miscellaneous systems are given in Table 2-4. The subsystem weights shown for the synchronous mission reflect the fail-operational/fail-operational/fail-safe design philosophy currently being considered in the orbit-to-orbit shuttle preliminary designs.

The stage geometry selected as the baseline for this analysis was the tandem tank configuration (10111). This geometry was selected primarily as a matter of convenience because the structure conversion factors (complex-to-monocoque structure weight ratios) used in the sizing program were more accurate for this configuration than for others. A typical stage geometry is illustrated in figure 2-1.

The costs, excluding the engine costs which were input, generated during this analysis were based on cost estimating relationships which are predicated on historical cost data and pertinent vehicle parameters.(4) In general, the cost estimating relationships of any cost element contain coefficients which indicate the technology level and complexity of that individual element. Table 2-5 presents the technology base assumed for the main systems on the stage. Table 2-6 lists the percent learning curves used to compute the investment costs.

The program cost data developed during this analysis include only the RDT&E, investment, and upper stage propellant costs. The program costs presented do not contain the other cost elements normally included in the operations costs, because they are mainly launch vehicle and mission model oriented, and for identical missions and similar size upper stages the operational costs are relatively insensitive to variations in upper stage weight. Hence, any program cost sensitivities which are determined from these program cost data will be of sufficient accuracy.

2.2.2 Synchronous Mission

2.2.2.1 Mission Profile

The profile selected for the synchronous mission was the transfer of payloads between a low inclination, low altitude, earth (parking) orbit and a synchronous orbit. This mission would require the liquid hydrogen-liquid oxygen stage to perform one of the following maneuvers:

1. Delivery of a payload from low earth orbit to synchronous orbit and return without a payload.

2. Transport a payload from low-parking orbit to synchronous orbit and return with the same or a different payload; and

Table 2-1. Summary of Stage Design Constraints

Constraint / Mission	Single Stage Synchronous	Interplanetary (Mars)
Maximum Stage Diameter (In.)	260	260
Shell - Tank Spacing (In.)	9.0	9.0
Tank - Tank Spacing (In.)	9.0	9.0
Engine - Tank Spacing Factor (Chamber)	4.0	4.0
Engine - Tank Spacing Factor (Exit)	0.8	0.8
Engine - Booster Spacing (In.)	0.0	0.0
Engine Gimbal Angle (Degrees)	3.0	3.0
Thrust - To - Weight Ratio	0.25	0.25
Axial Acceleration (G's)	1.00	1.00
Lateral Acceleration (G's)	0.05	0.05
Payload Density (Lb/Ft^3)	25.0	25.0
Inert Weight Contingency Factor (%)	7.5	7.5

Table 2-2. Summary of Structural Design Data

Data \ Structure	Shell	Thrust Cone
Material	Aluminum	Aluminum
Density (lb/ft^3)	183.0	183.0
Material Strength (psi)		
Tension	67,000	67,000
Compression	46,000	46,000
Modulus of Elasticity (psi)	10^7	10^7
Safety Factors		
Tension	1.25	1.25
Compression	1.00	1.00
Monocoque-to-Complex Structure Weight Ratio	*	*
Spider Beam Multiplication Factor	N/A	N/A

* A function of diameter and limit load; see appendix C.

Table 2-3. Summary of Tankage Design Data

Data \ Mission	Synchronous	Interplanetary (Mars)
Tankage		
Material	Aluminum	Aluminum
Density (lb/ft^3)	183.0	183.0
Allowable Stress (psi)	60,000	60,000
Factor of Safety	1.10	1.10
Minimum Skin Gauge (In.)	0.025	0.025
Land Factors (Bulkheads)	0.10	0.10
Land Factors (Cylindrical Section)	0.05	0.05
Thermal Protection		
Initial Fuel/Oxidizer Temperature (^oR)	36.0/162.6	36.0/162.6
Initial Fuel/Oxidizer Pressure (psi)	15.0/15.0	15.0/15.0
External Insulation Temperature (^oR)	450.0/470.0	400.0/420.0
Insulation Density (lb/ft^3)	4.5	4.5
Insulation Thermal Conductivity (Btu/Hr-Ft-^oR)	*	*
Meteoroid Protection		
Probability of no Punctures	N/A	0.995
Nominal Mission Altitude (n.m.)	N/A	500,000
Shield Material	N/A	Aluminum
Material Density (lb/ft^3)	N/A	183.0
Material Yield Stress (psi)	N/A	70,000
Minimum Skin Gauges (In.)	N/A	0.015
Miscellaneous		
Minimum Fuel/Oxidizer Ullage Volume (%)	5.0/5.0	5.0/5.0
Residual Fuel/Oxidizer Fraction (%)	2.0/2.0	2.0/2.0
Feedline Flow Velocity (fps)	20.0	20.0
Tank Support Factor	ξ	ξ

* A function of temperature and thickness; see appendix C.
ξ Dependent upon configuration; see appendix C.

Table 2-4. Miscellaneous Subsystem Weights (Pounds)

Subsystem \ Mission	Synchronous	Interplanetary (Mars)
Hydraulic/Pneumatic	550	60
Destruct	50	20
Propellant Utilization	500	50
Communications	400	110
Instrumentation	200	50
Guidance, Navigation and Control	600	250
Electrical	200	30
Electric Power and Distribution	2000	110
Total	4500	680

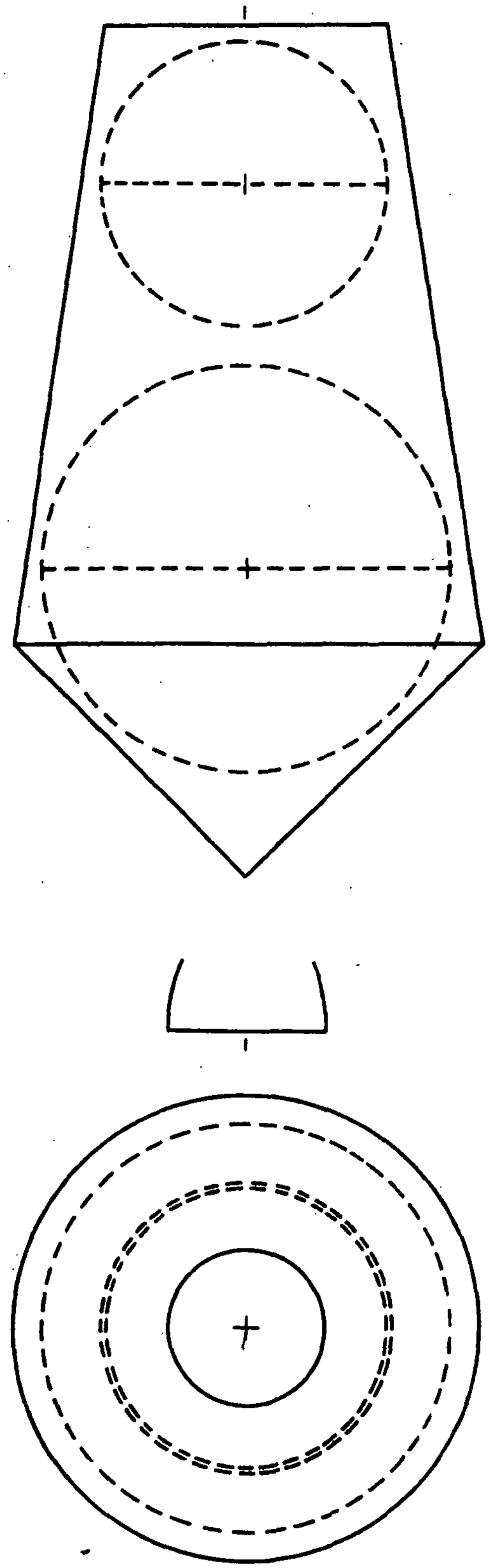

Figure 2-1. Tandem Tank Stage Configuration (10111)

Table 2-5. Technology Level of Systems

Area	Technology Level and Technique*
Structures	
Shell	SOA - Aluminum Sheet Stringer
Thrust Structure	SOA - Aluminum Sheet Stringer
Tankage	SOA - Aluminum Monocoque
Meteoroid Shield	SOA - Aluminum Monocoque
Tank Support	ADV - Composite
Propellant Feedlines	SOA - Aluminum
Propulsion	
Main Engines	N/A - Cost Input
Reaction Control Thrusters	SOA - Monopropellant
Miscellaneous Subsystems	Adaptation of existing hardware to a new unmanned, reuseable upper stage
Electrical Power and Distribution	
Electrical	
Communication	
Instrumentation	
Guidance, Navigation and Control	
Hydraulic/Pneumatic	
Propellant Utilization	
Destruct	

*SOA - State-of-the-art Technology
ADV - Advance Technology

Table 2-6. Investment Learning Curves

System	Learning Curve
Structures	90%
Propulsion	95%
Miscellaneous Subsystems	90%
Assembly and Checkout	90%

3. Fly empty from the parking orbit to synchronous orbit, and return with a payload to the original departure orbit.

The typical profile for this mission is depicted in figure 2-2. This involves a Hohmann-type transfer maneuver from low earth orbit to synchronous altitude, a plane change and circularization at synchronous orbit, a return Hohmann transfer and plane change at synchronous orbit and circularization into the original low earth orbit.

The four velocities used for this mission assumed a Hohmann-type transfer between a 28½-degree inclination, 100-nautical-mile circular orbit and an equational (0-degree inclination) synchronous orbit. The velocities were corrected to account for the effects of the stage's initial thrust-to-weight ratio and specific impulse. However, the effect of orbital regression on the velocity requirements was not considered.

2.2.2.2 Stage Weights for the Synchronous Mission

Stages were analyzed with payloads which ranged from 0-pound round trip to 15,000-pound round trip, and various combinations of specific impulses and engine weights. Specific impulses of 455,465 and 475 seconds, and engine weights of 100, 1000, and 3500 pounds were selected to ensure that most, if not all, possible combinations were included. The results of the stage sizing analyses are presented as carpet plots in figures 2-3 through 2-11. These nine figures show the stage weights for different sets of payload up and payload down, and various combinations of engine specific impulse and weight.

2.2.2.3 Cost Data for the Synchronous Mission

RDT&E, Theoretical First Unit (TFU), and program costs were determined for the various stages sized in this analysis (section 2.2.2.2). The costs were computed with a computer routine, developed specifically for upper stages, which determines costs from cost estimating relationships.[4] However, in this analysis the various combinations of engine RDT&E and TFU costs were input to the program to ensure that the results would be applicable to a wide range of developments in engine technology. Engine RDT&E costs of 50.0, 225.0 and 400.0 million dollars, and TFU engine costs of 0.5, 1.0 and 1.5 million dollars were selected for this analysis. The resulting costs are depicted in figures 2-12 through 2-22.

Stage RDT&E costs, independent of engine first unit costs, are presented for the 3-engine RDT&E costs as a function of stage weight.

Figure 2-13 shows the TFU stage costs for the 3-engine TFU costs as a function of stage weight. The TFU stage costs are independent of engine RDT&E costs.

The program costs for the synchronous mission are presented for the nine combinations of engine RDT&E and TFU costs as a function of stage weight. Although presented as program costs, the costs shown in these figures include only the RDT&E, investment, and upper stage propellant costs. They do not contain the other cost elements normally included in the operations costs, because they are mainly launch vehicle and mission model oriented, and for identical missions and similar size upper stages the operational costs are

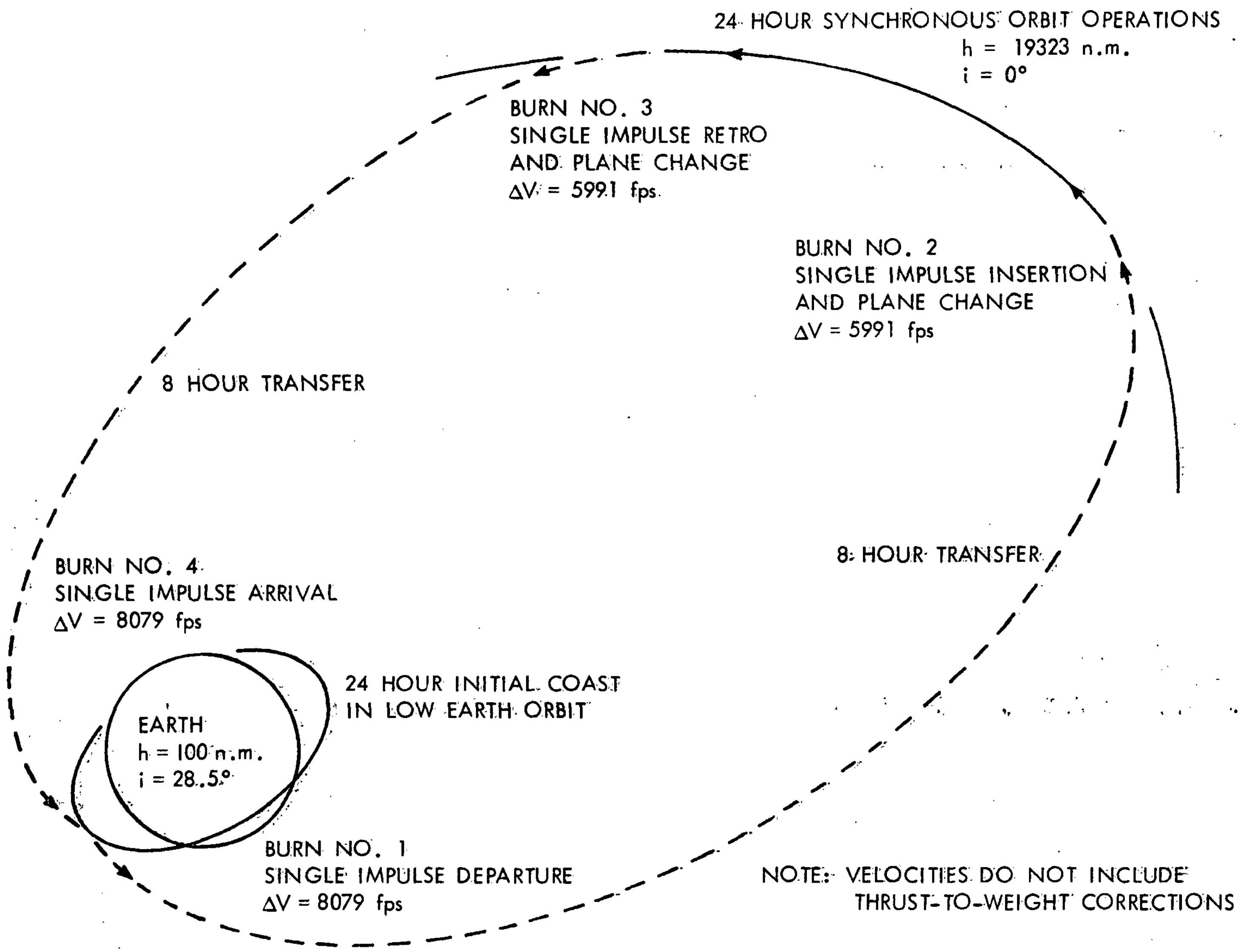

Figure 2-2. The Synchronous Mission Profile

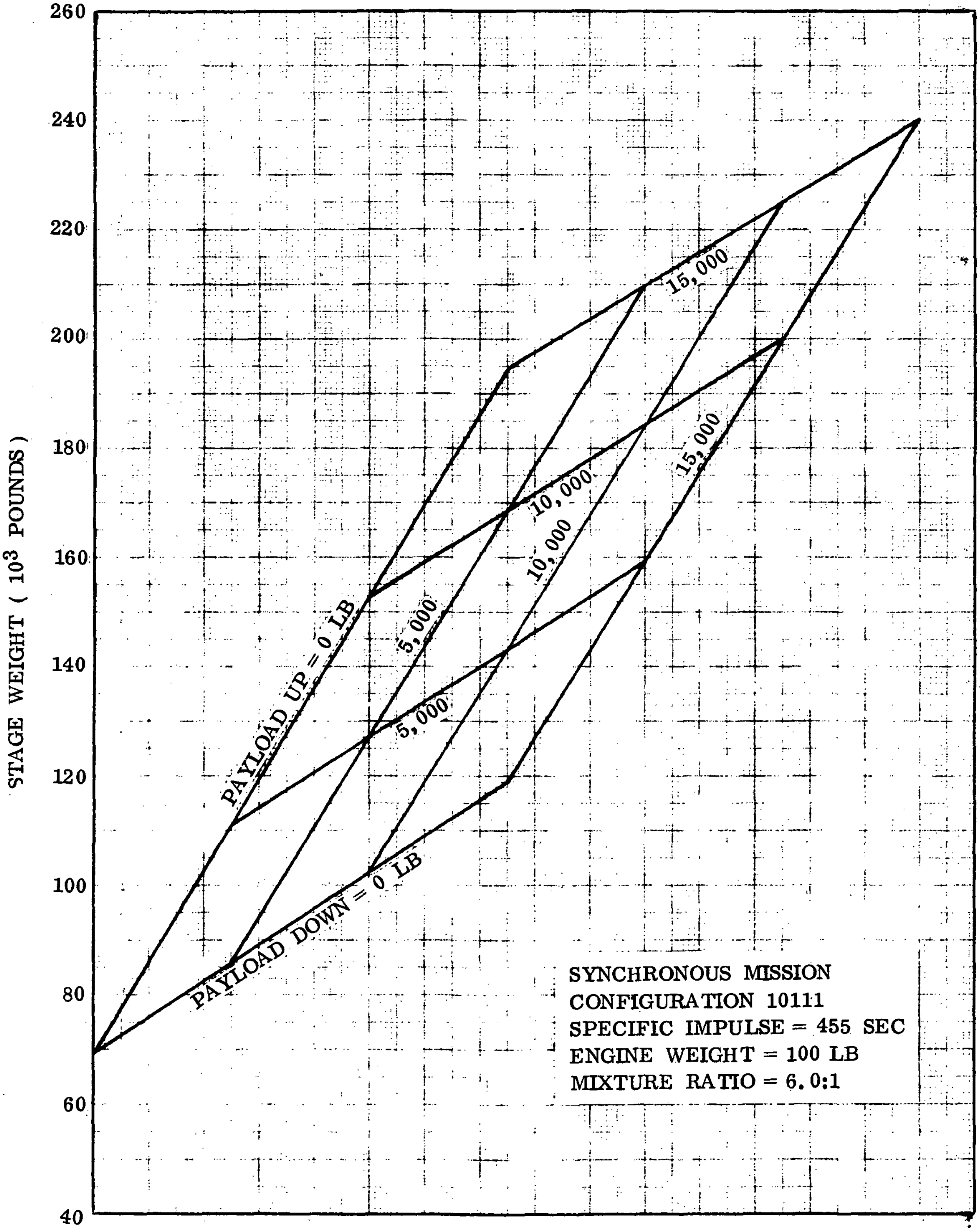

Figure 2-3. Synchronous Mission Stage Weights (Isp=455, Weng=100)

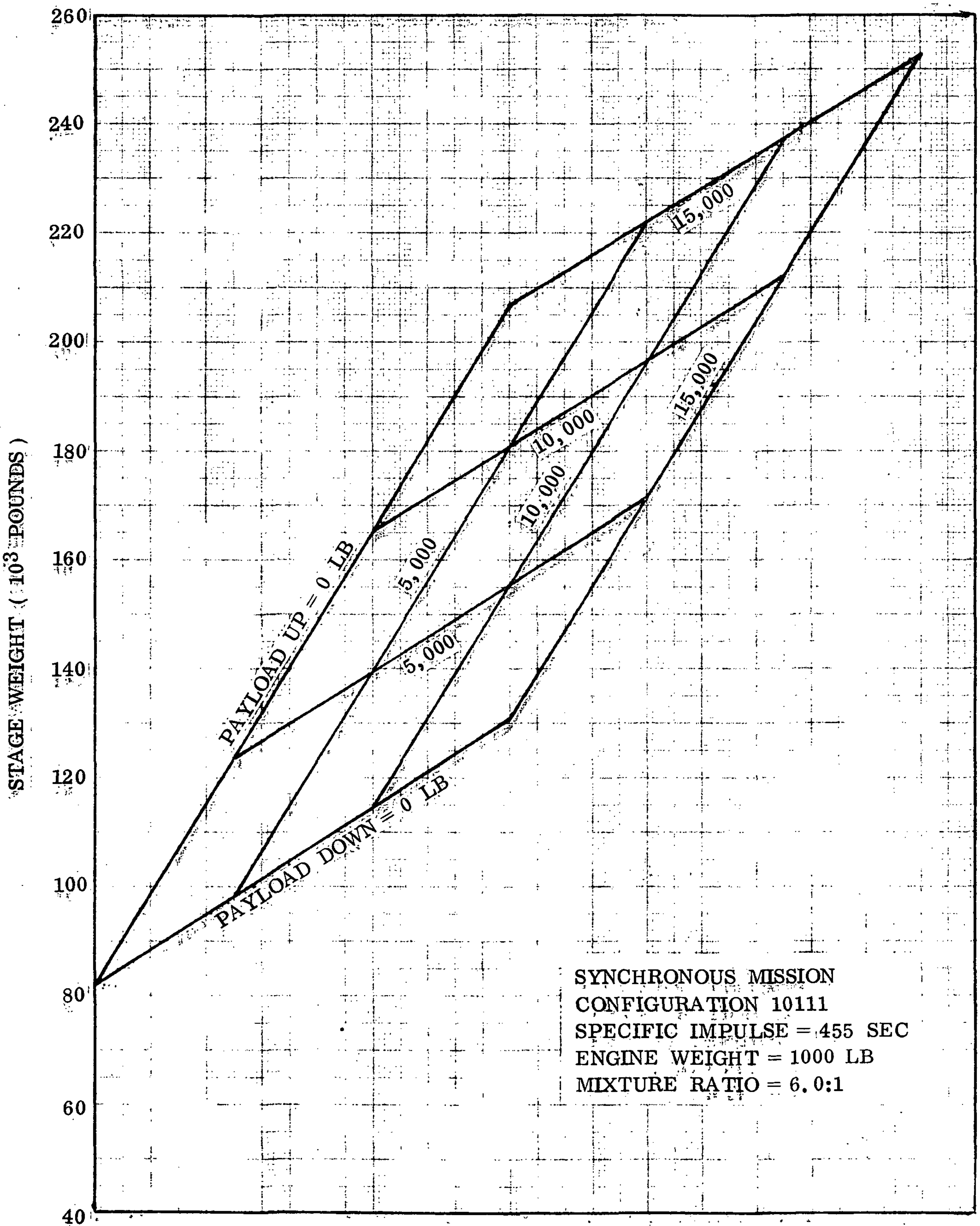

Figure 2-4. Synchronous Mission Stage Weights (Isp=455, Weng=1000)

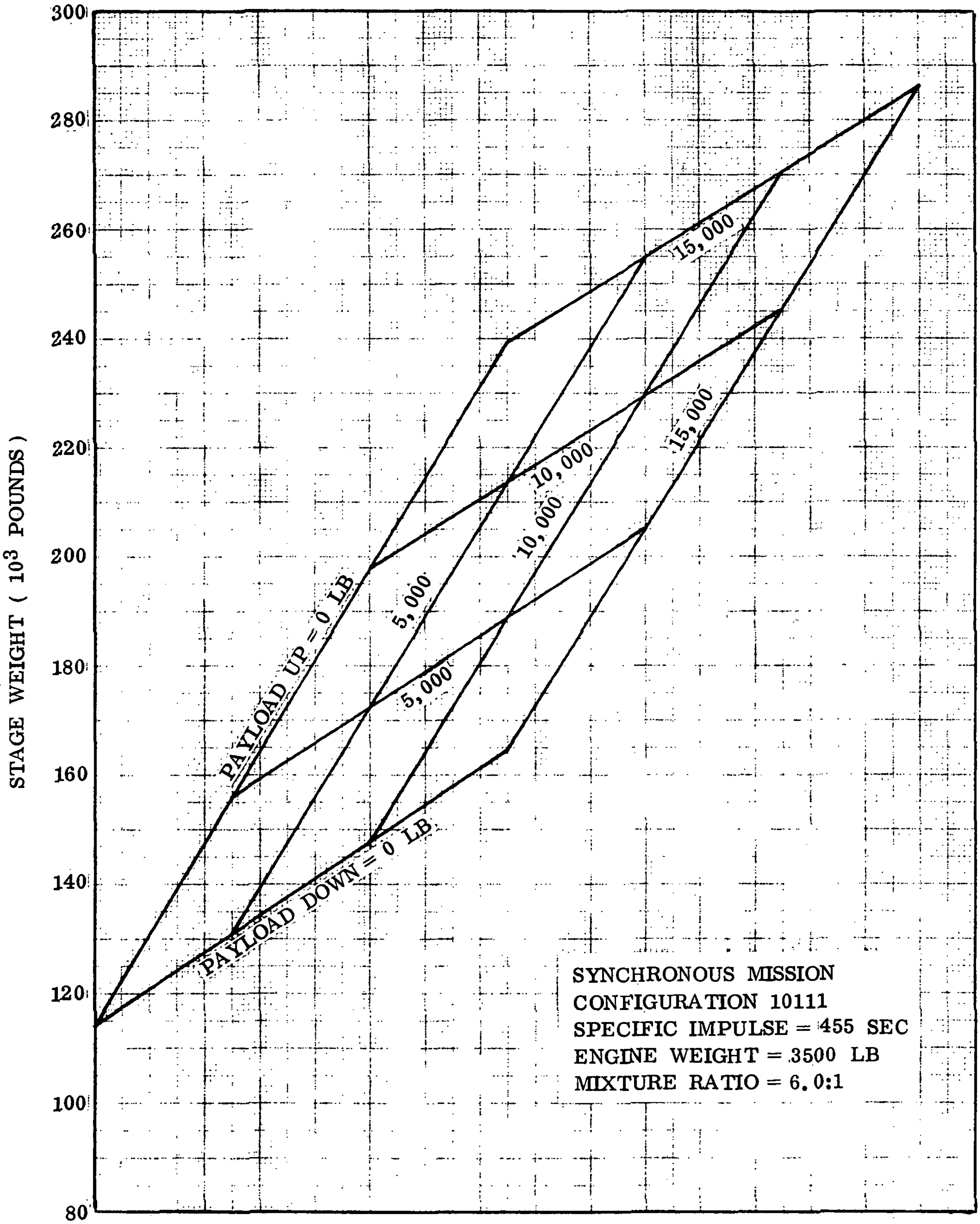

Figure 2-5. Synchronous Mission Stage Weights(Isp=455, Weng=3500)

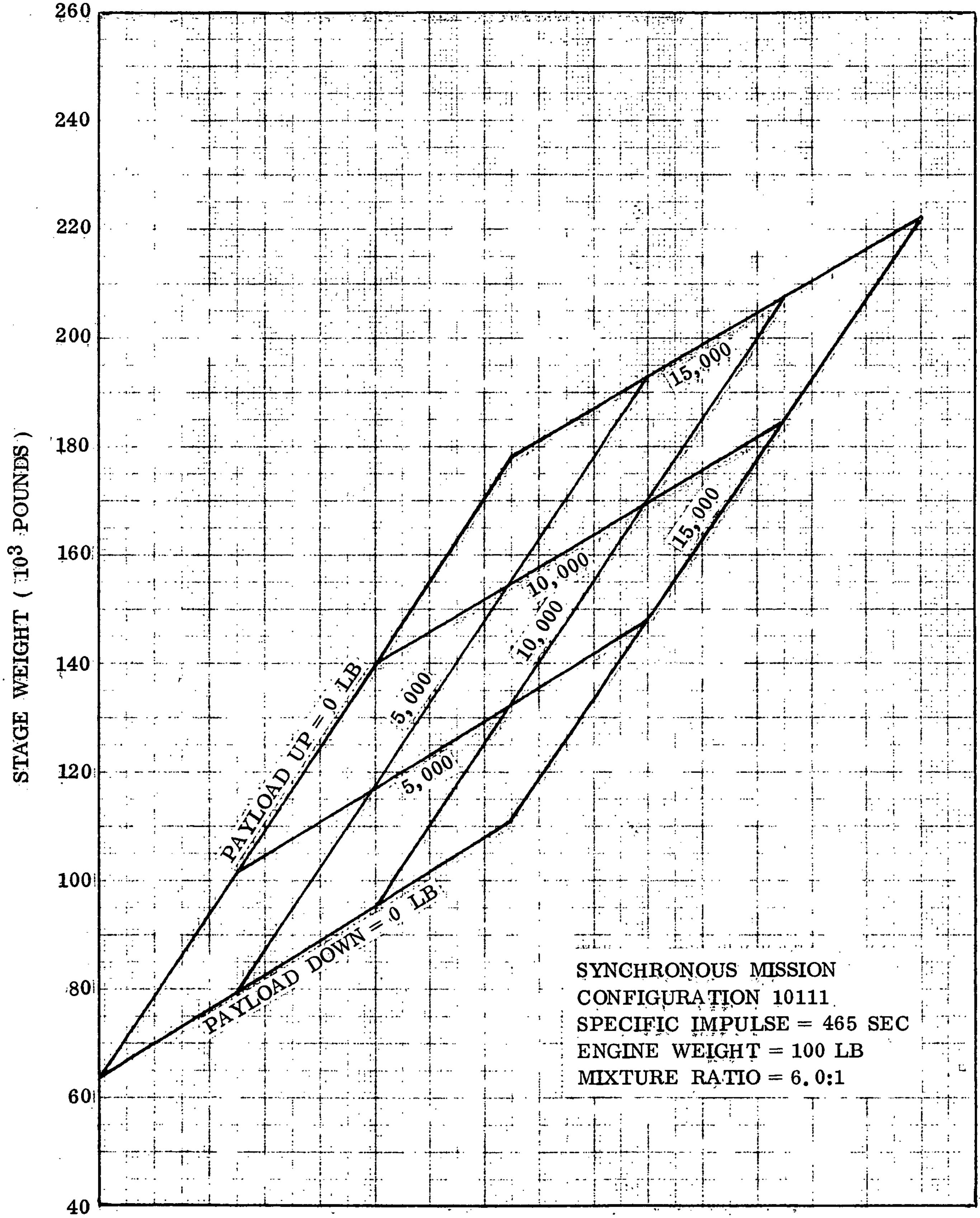

Figure 2-6. Synchronous Mission Stage Weights (Isp=465, Weng=100)

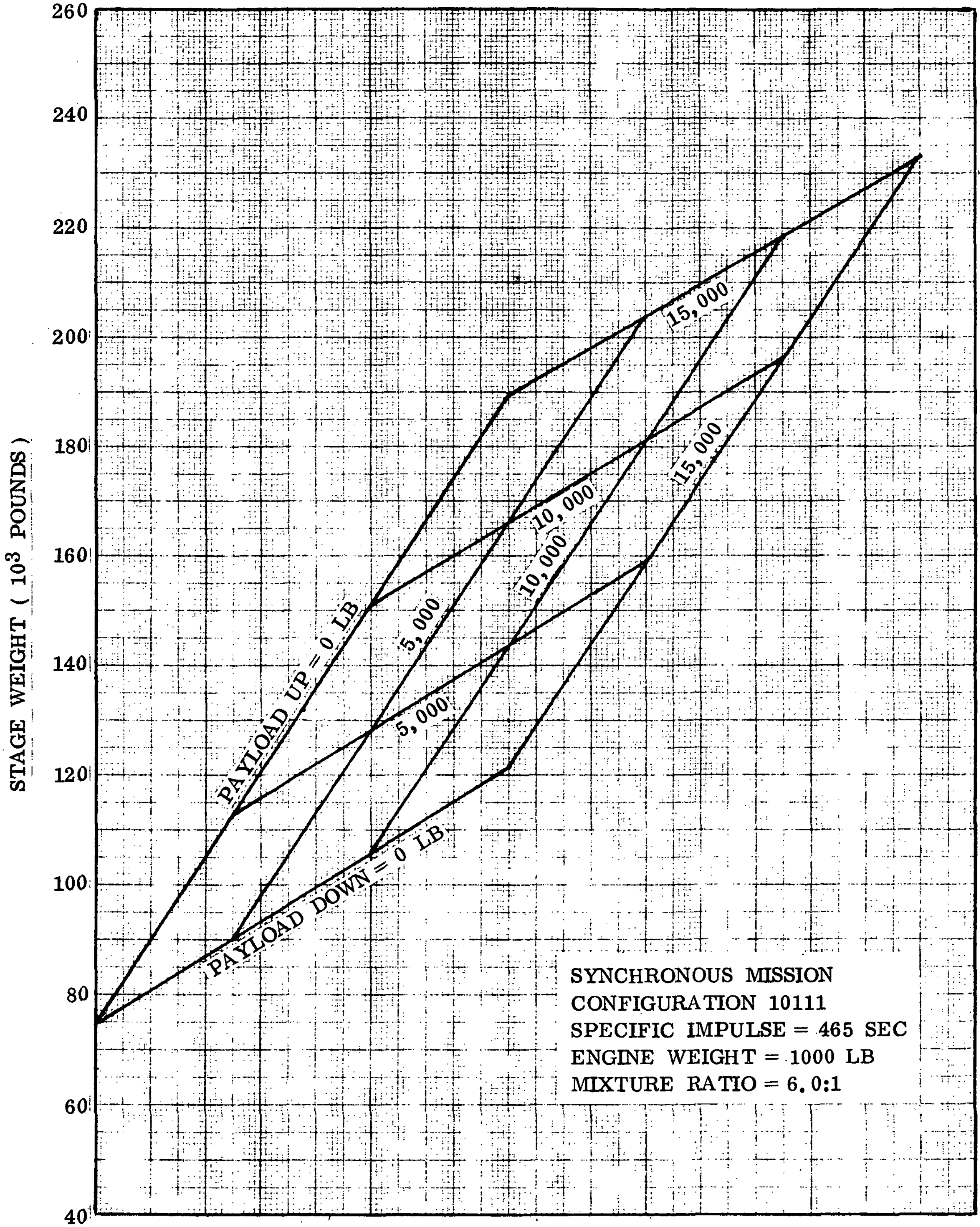

Figure 2-7. Synchronous Mission Stage Weights (Isp=465, Weng=1000)

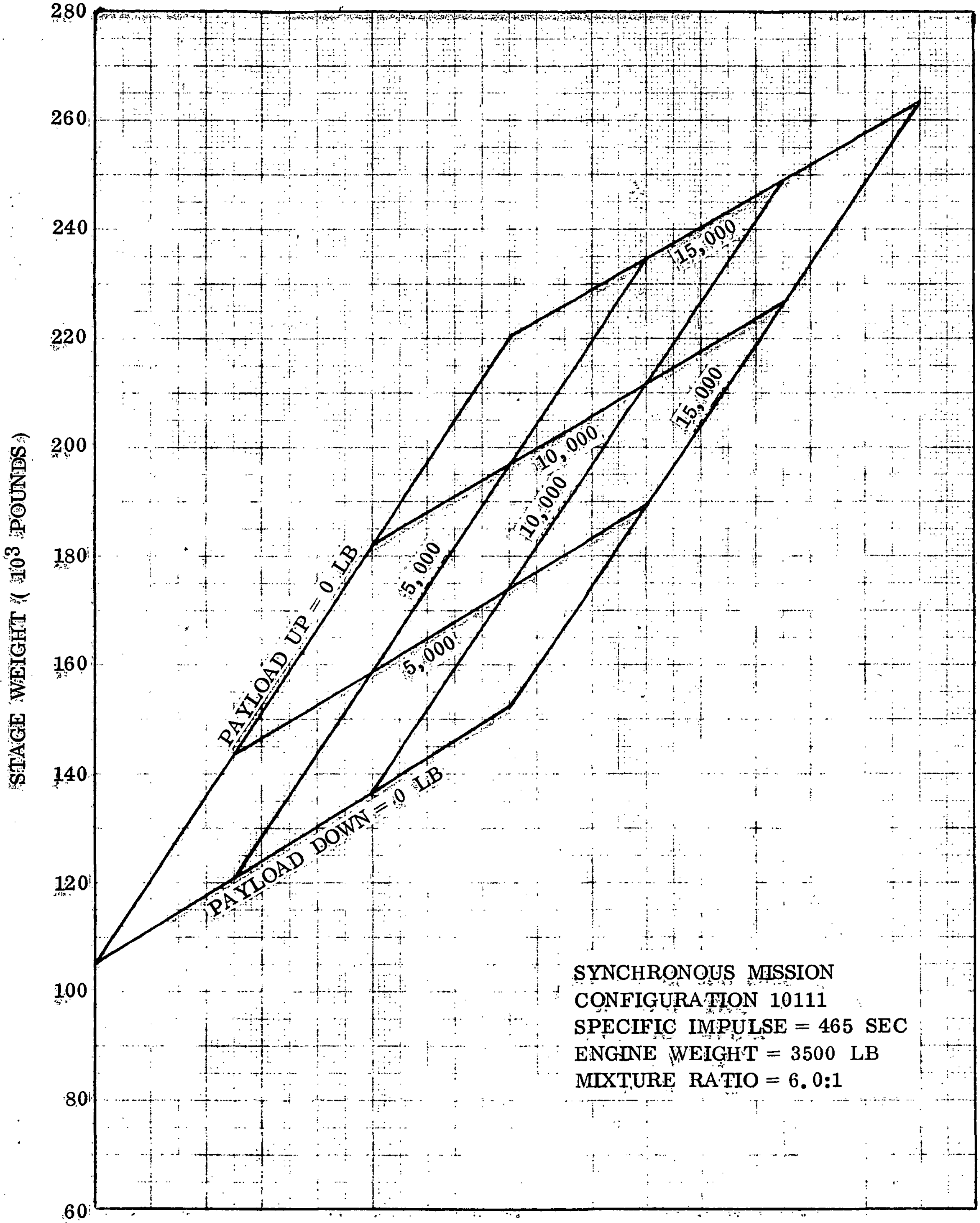

Figure 2-8. Synchronous Mission Stage Weights (Isp=465, Weng=3500)

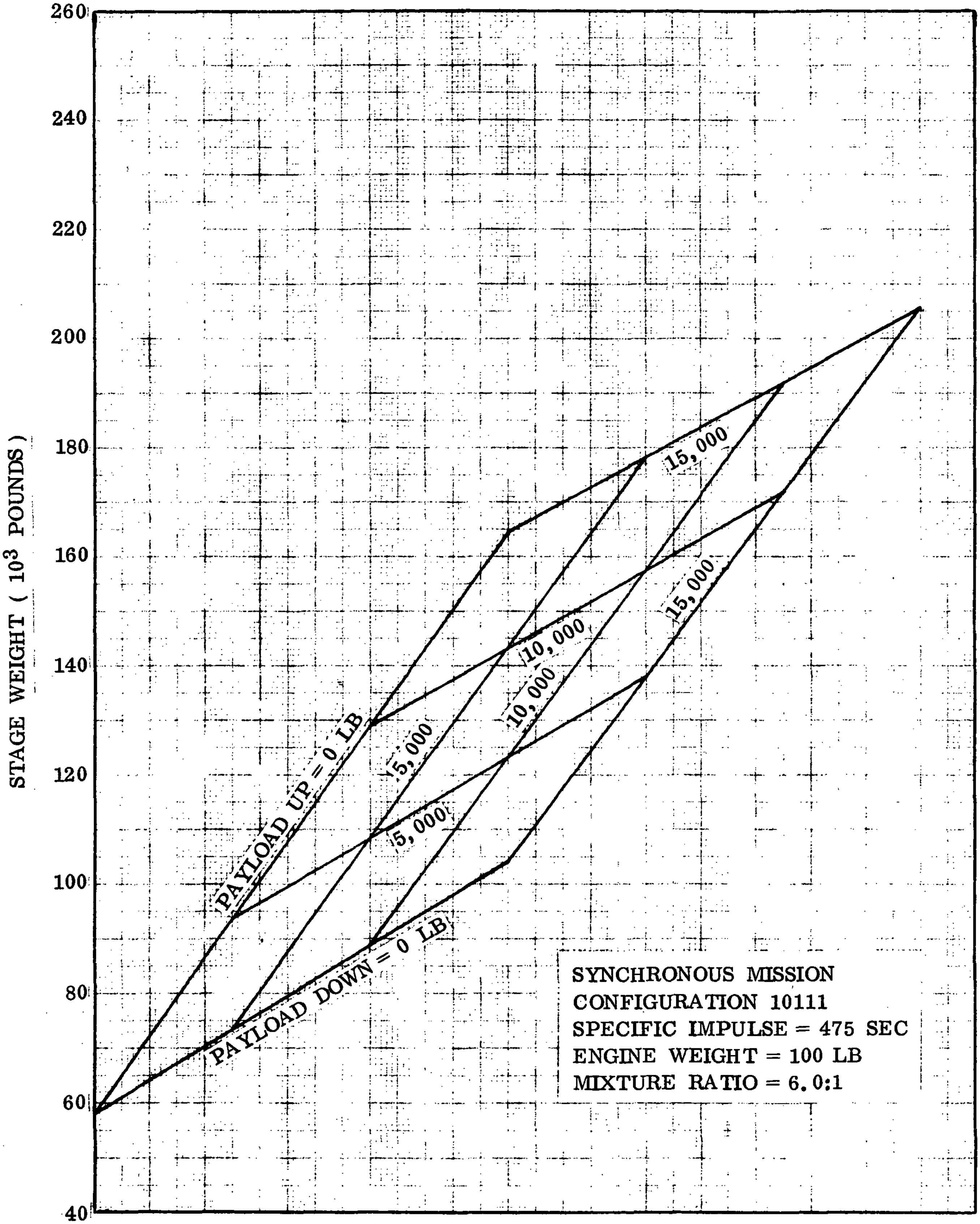

Figure 2-9. Synchronous Mission Stage Weights (Isp=475, Weng=100)

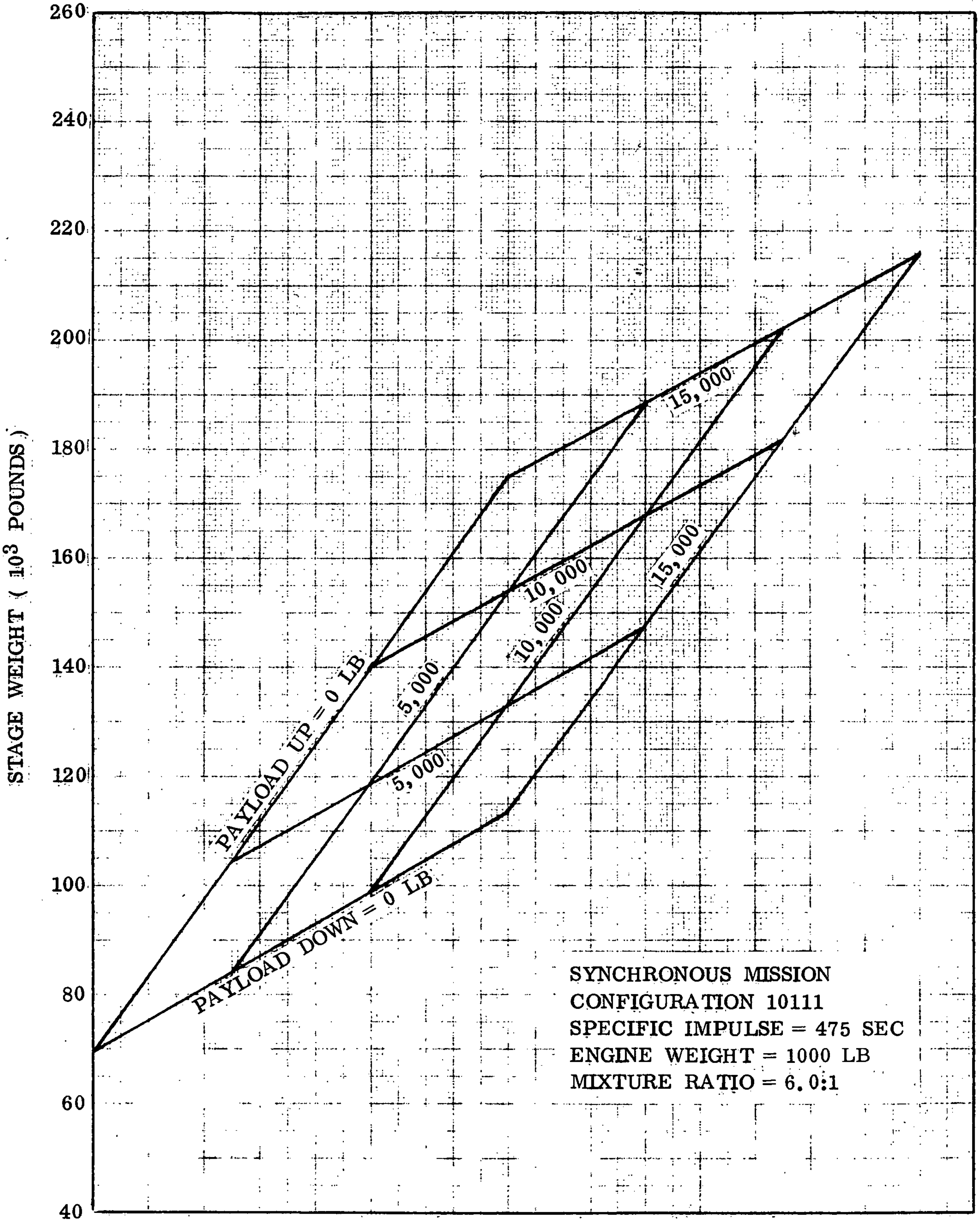

Figure 2-10. Synchronous Mission Stage Weights (Isp=475, Weng=1000)

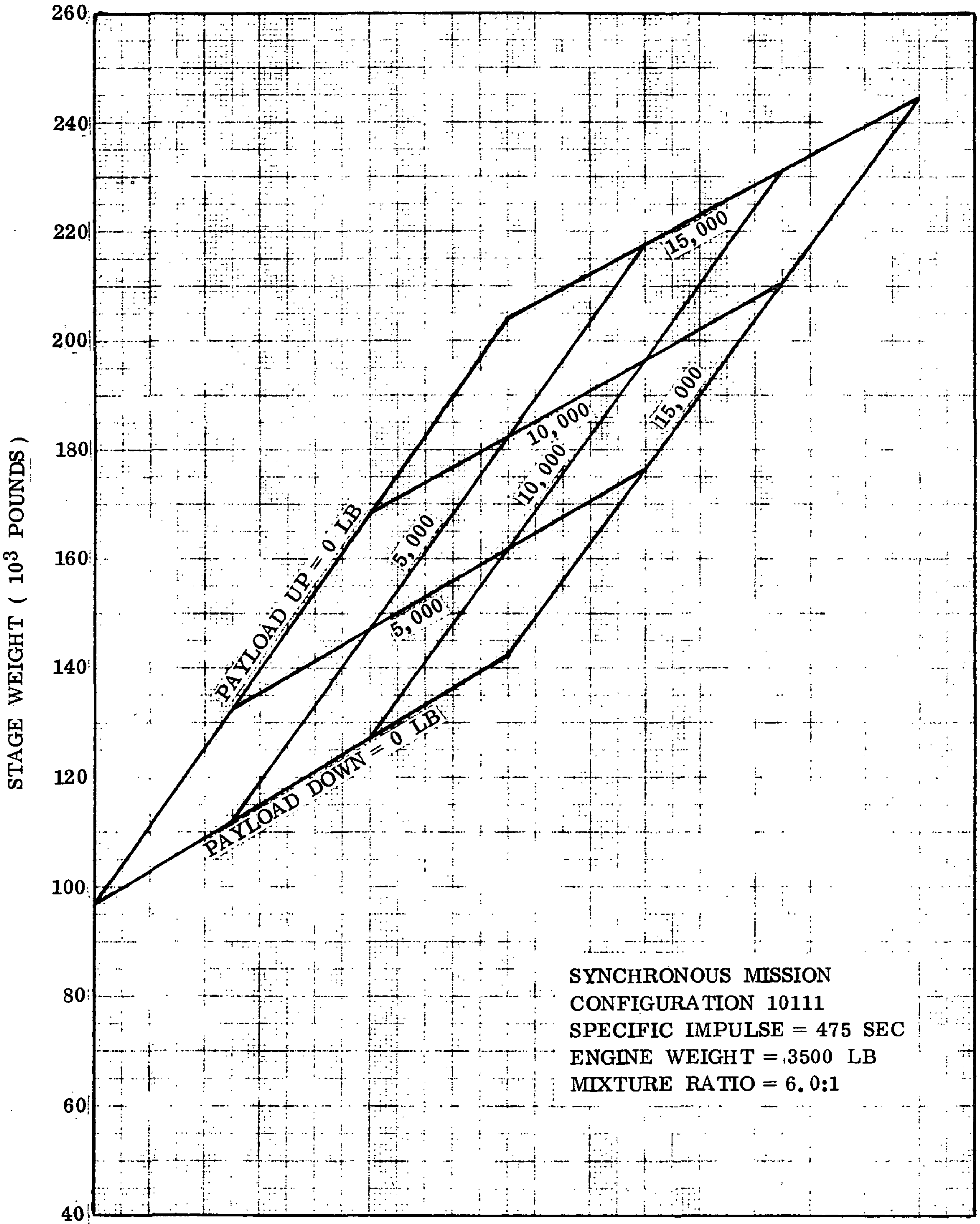

Figure 2-11. Synchronous Mission Stage Weights (Isp=475, Weng=3500)

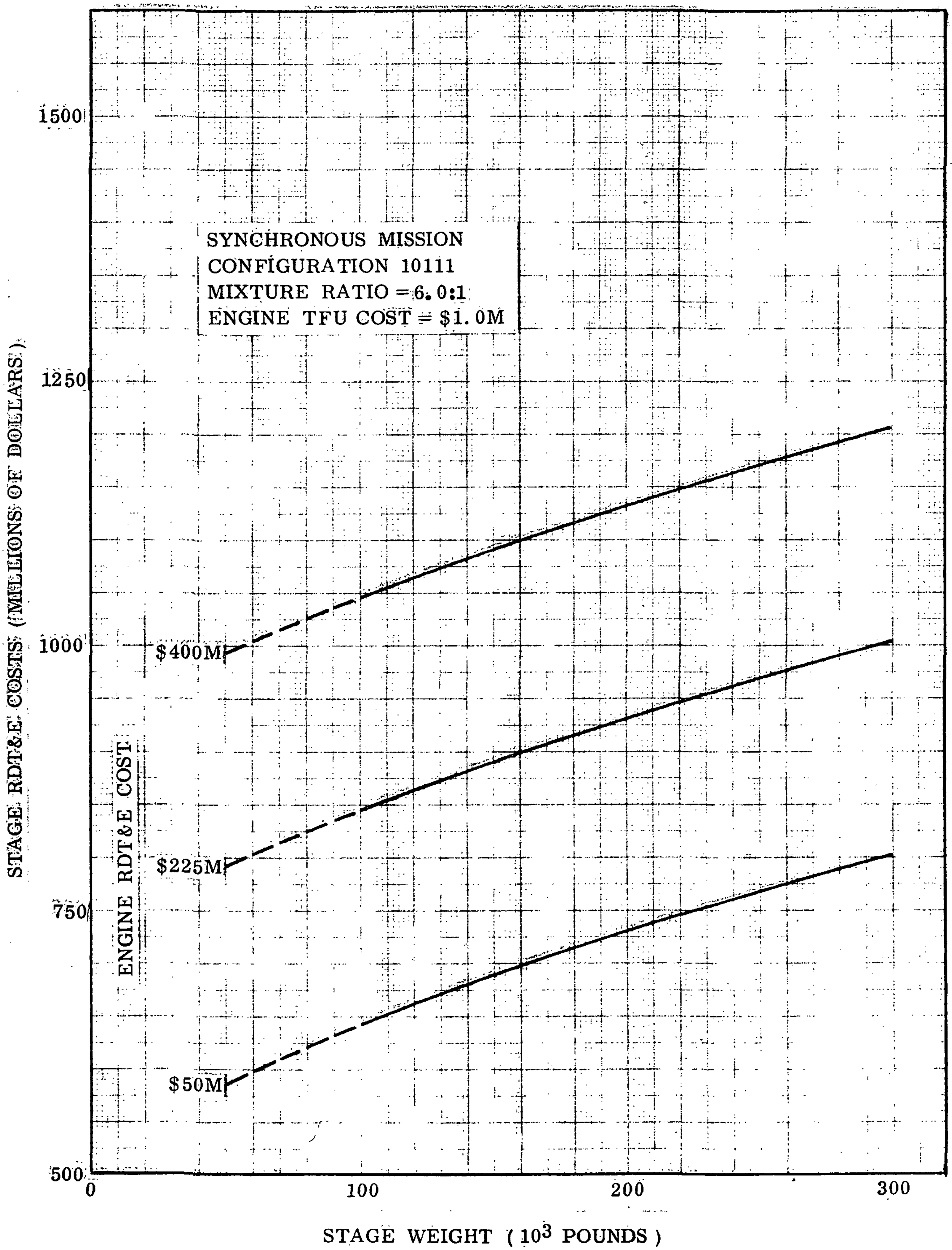

Figure 2-12. Stage RDT&E Costs for a Synchronous Mission

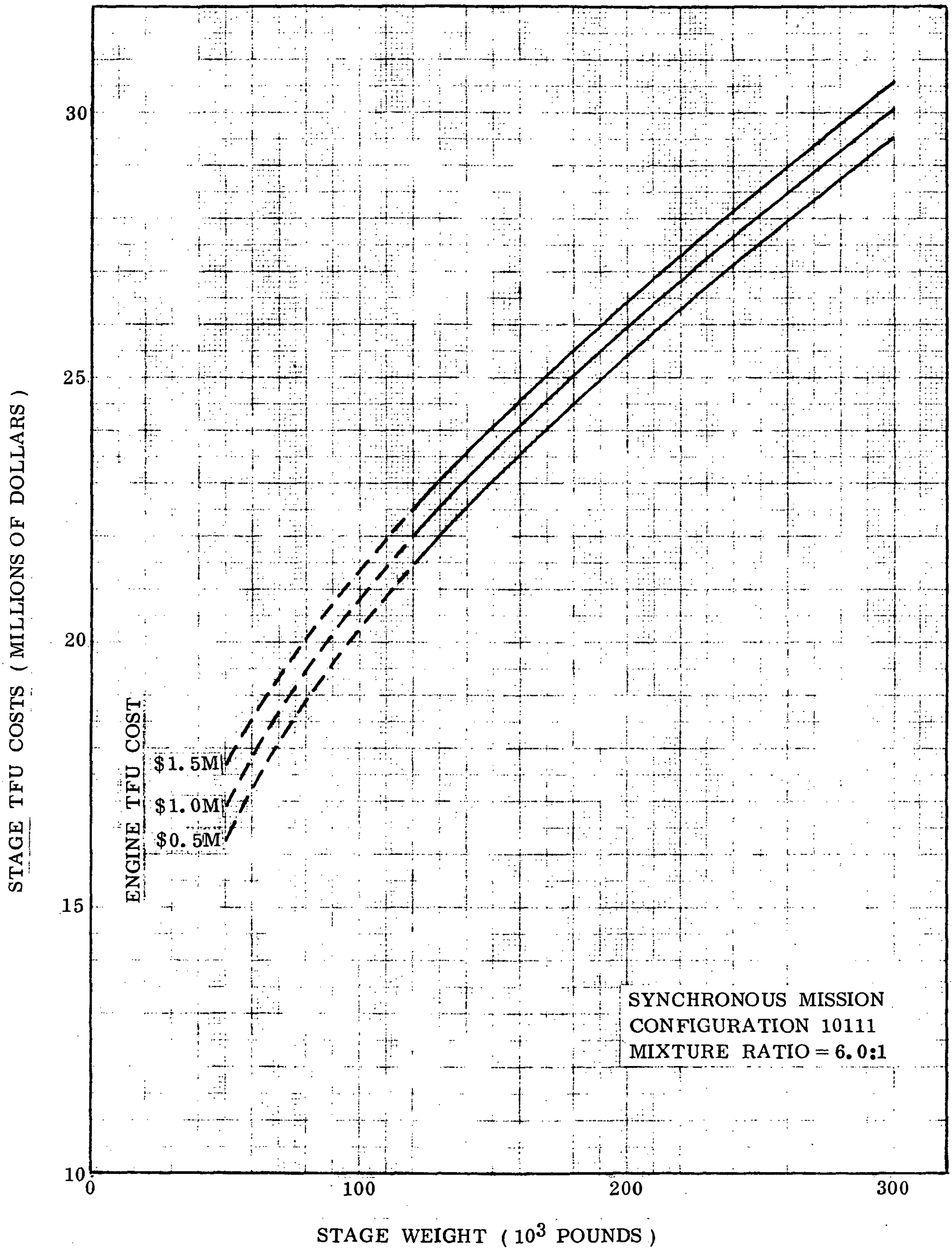

Figure 2-13. Stage First Unit Costs for a Synchronous Mission

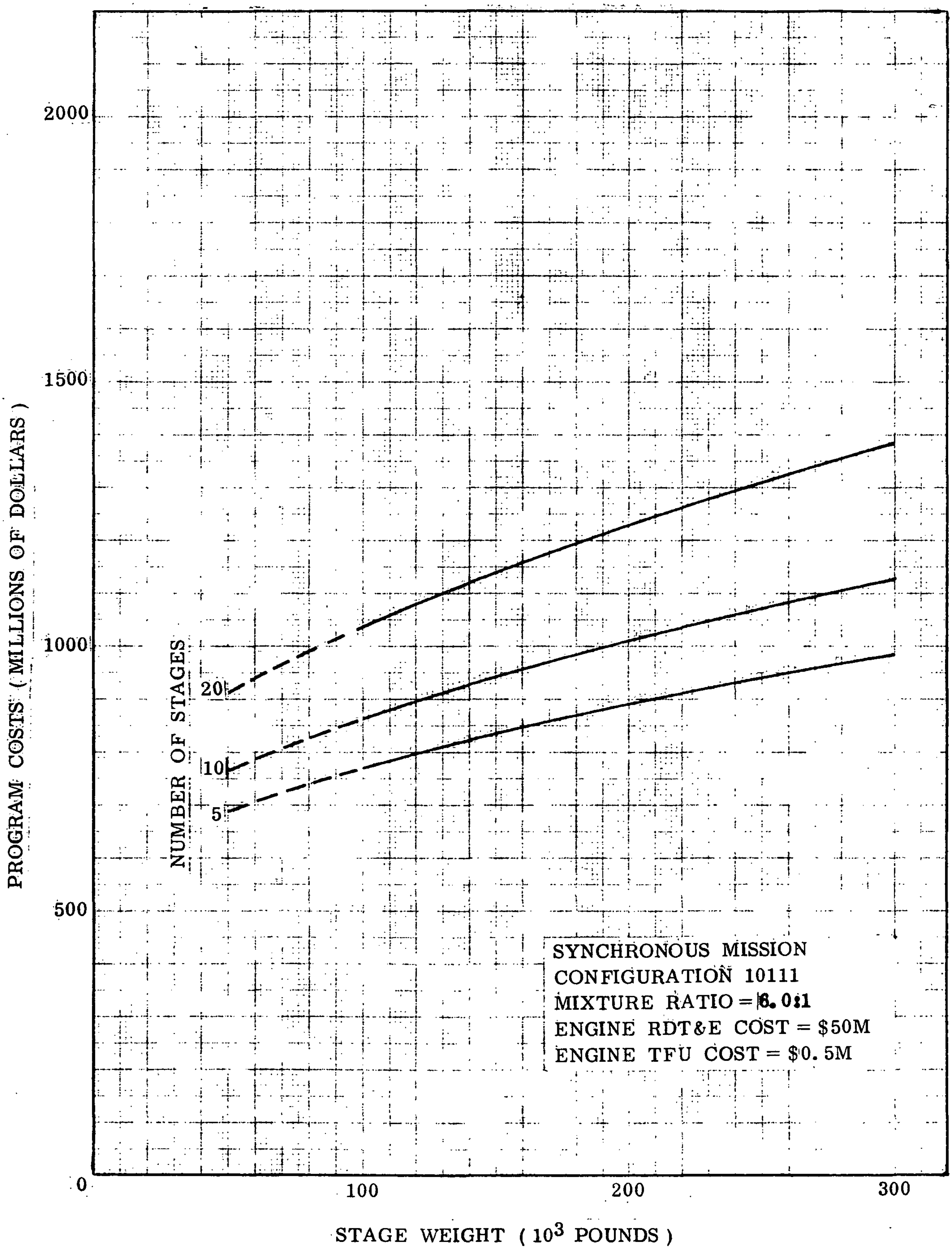

Figure 2-14. Program Cost for a Synchronous Mission (RDT&Eeng=$50M, TFUeng=$0.5M)

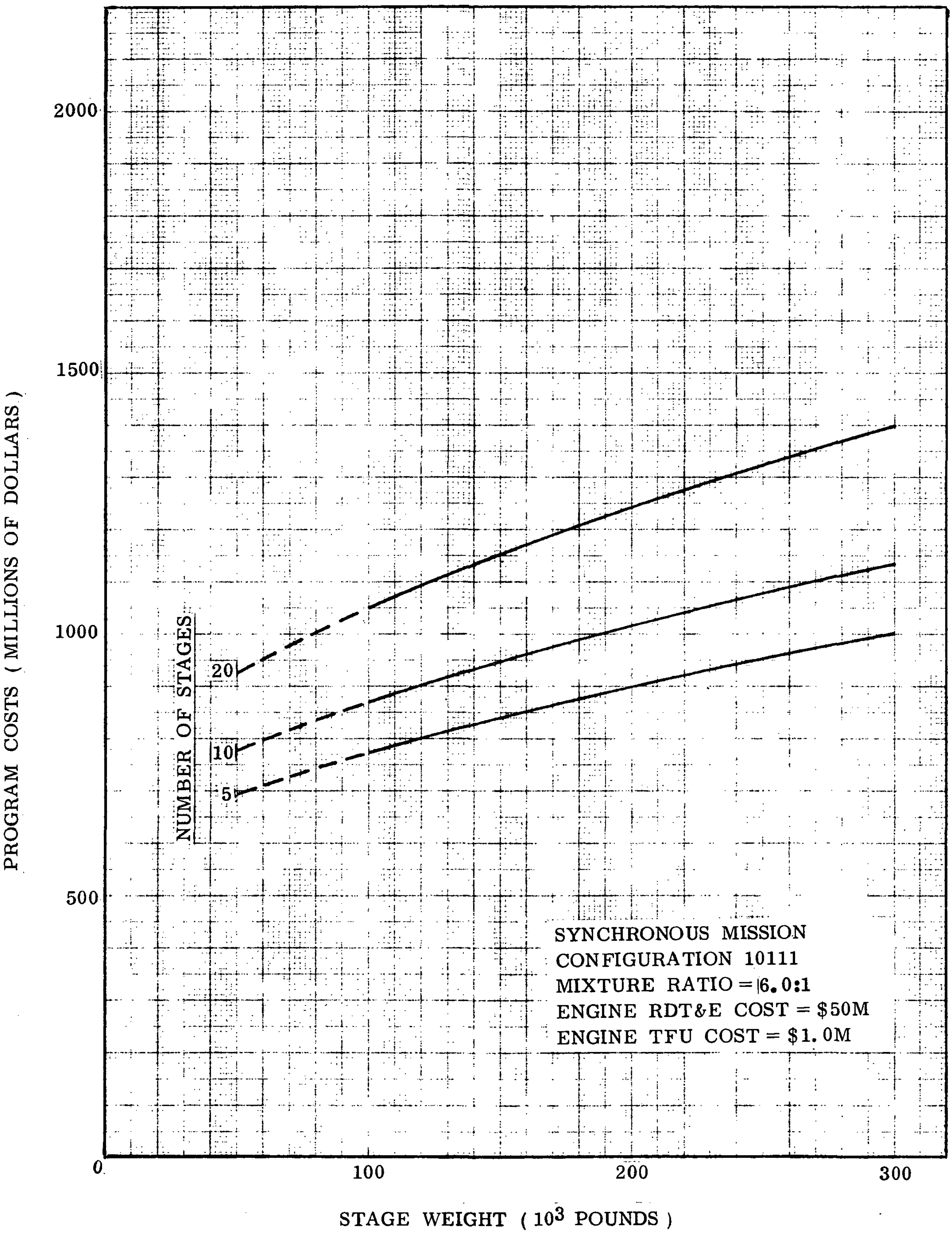

Figure 2-15. Program Costs for a Synchronous Mission (RDT&Eeng=$50M, TFUeng=$1.0M)

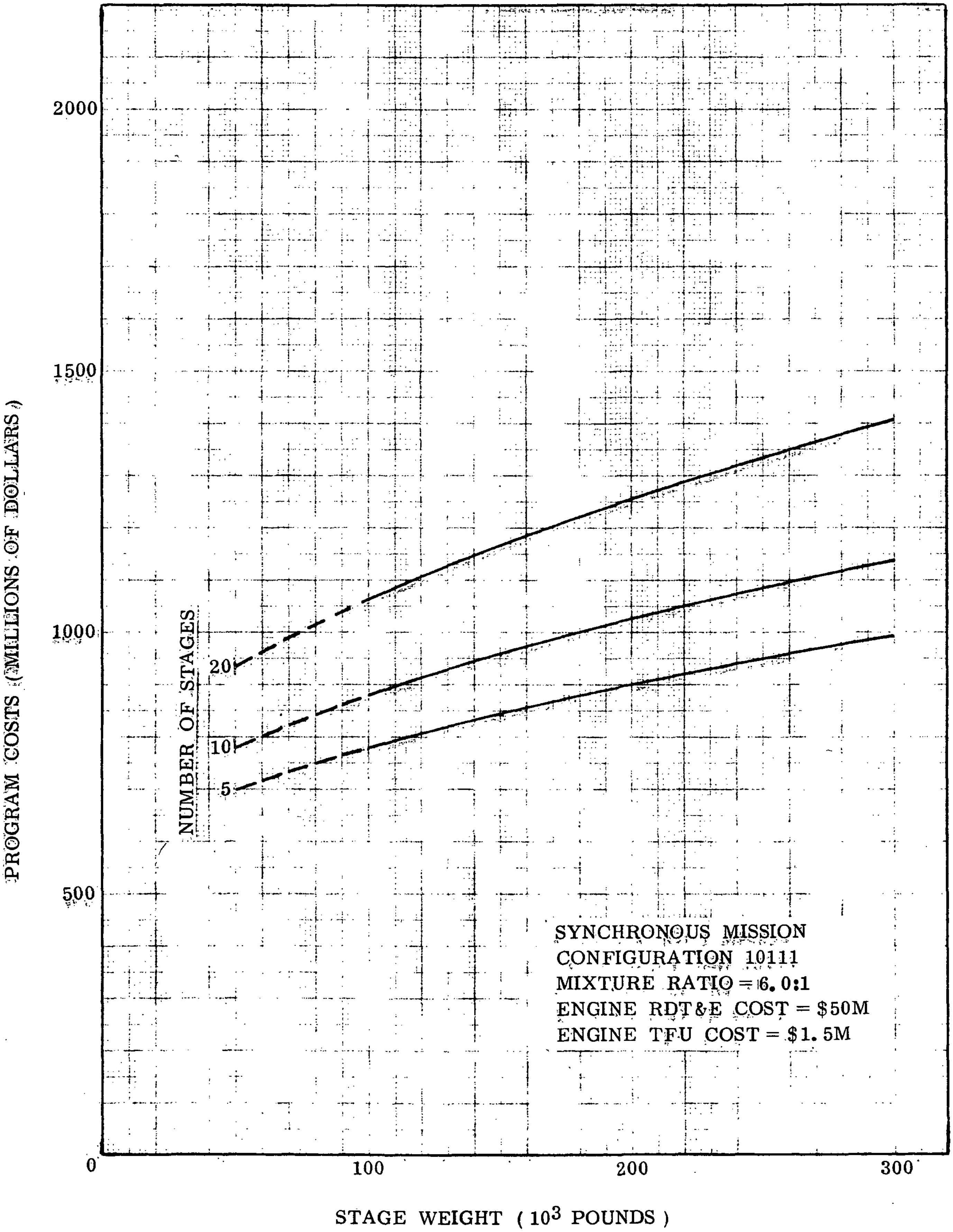

Figure 2-16. Program Costs for a Synchronous Mission (RDT&Eeng=$50M, TFUeng=$1.5M)

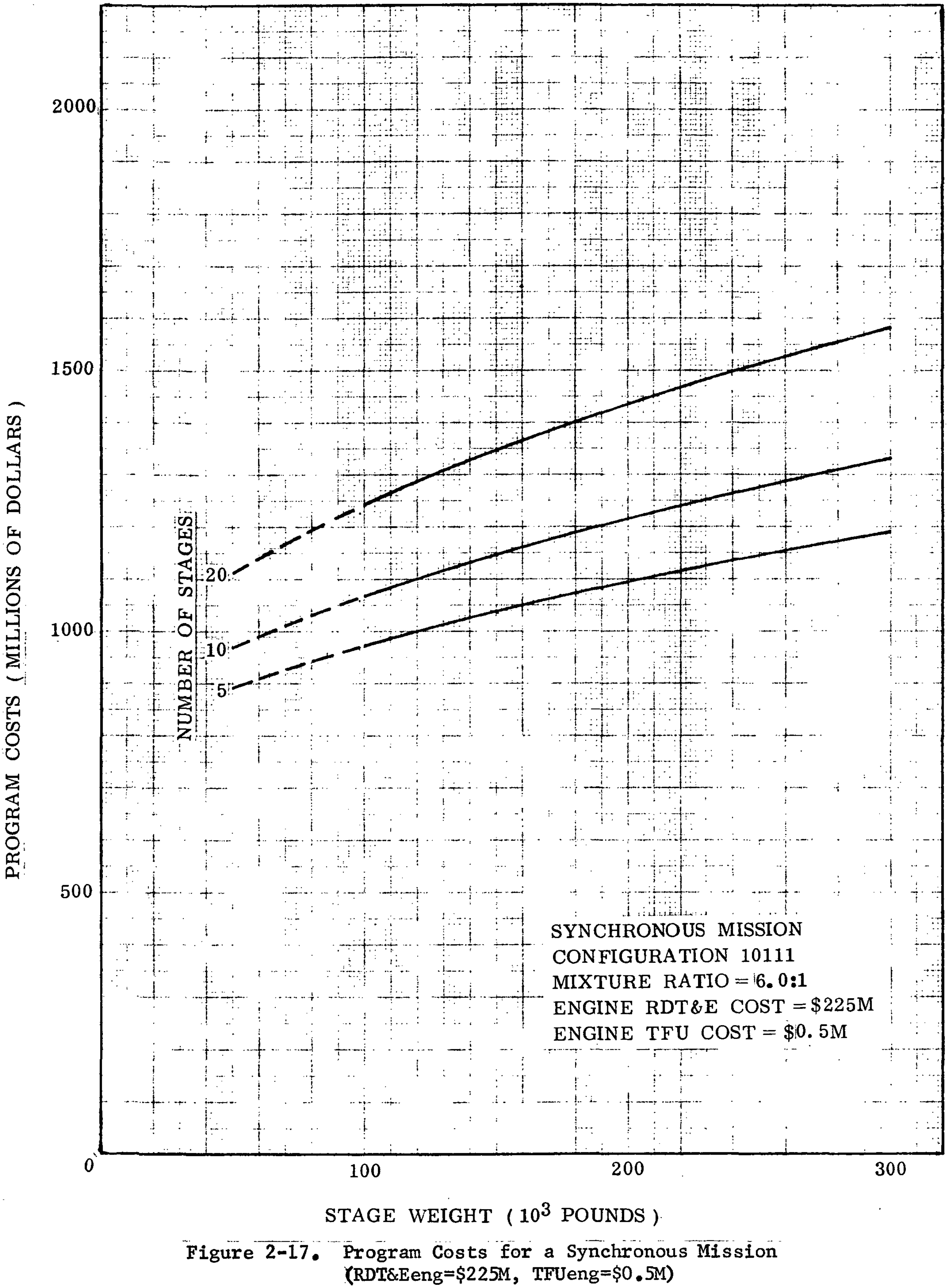

Figure 2-17. Program Costs for a Synchronous Mission (RDT&Eeng=$225M, TFUeng=$0.5M)

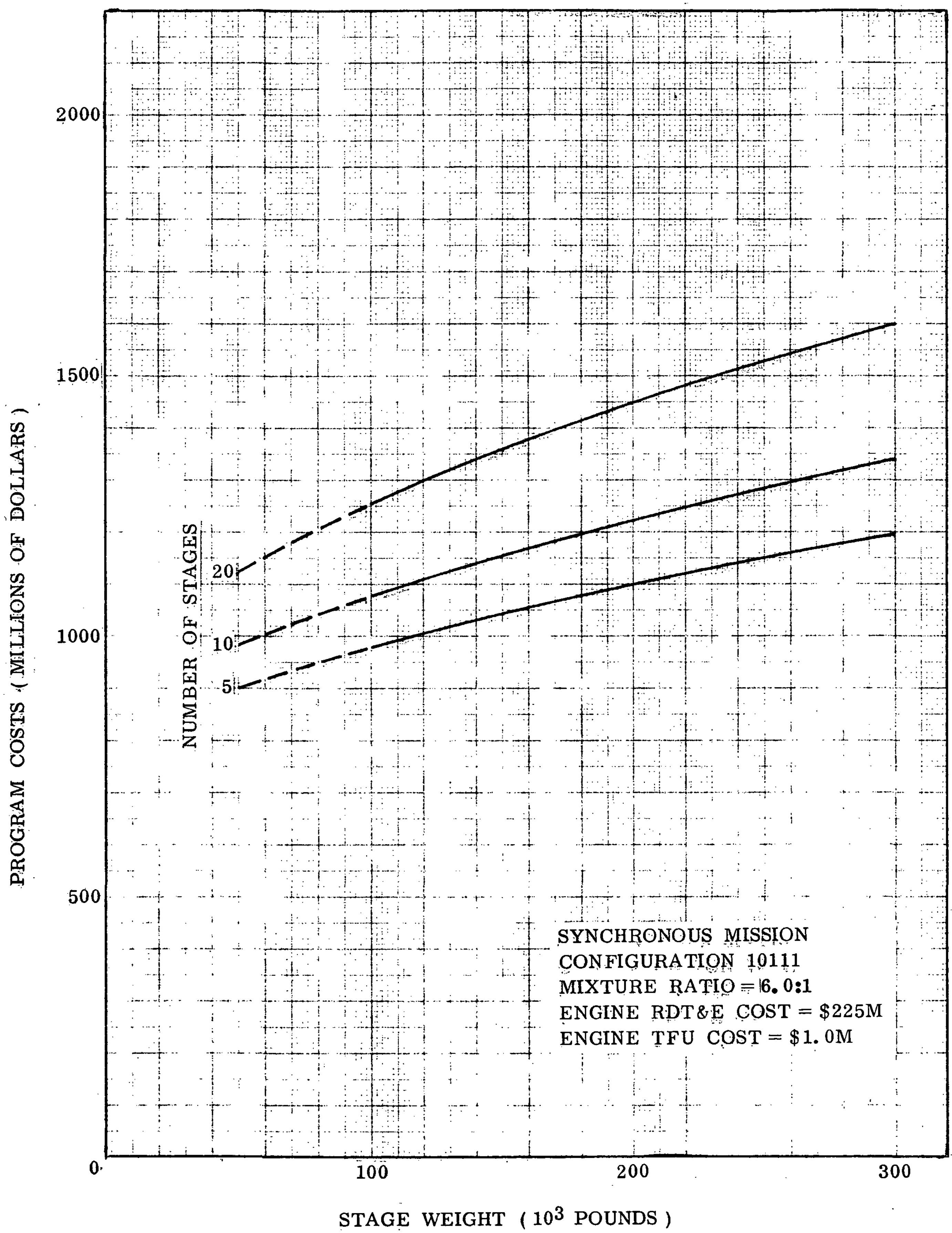

Figure 2-18. Program Costs for a Synchronous Mission (RDT&Eeng=$225M, TFUeng=$1.0M)

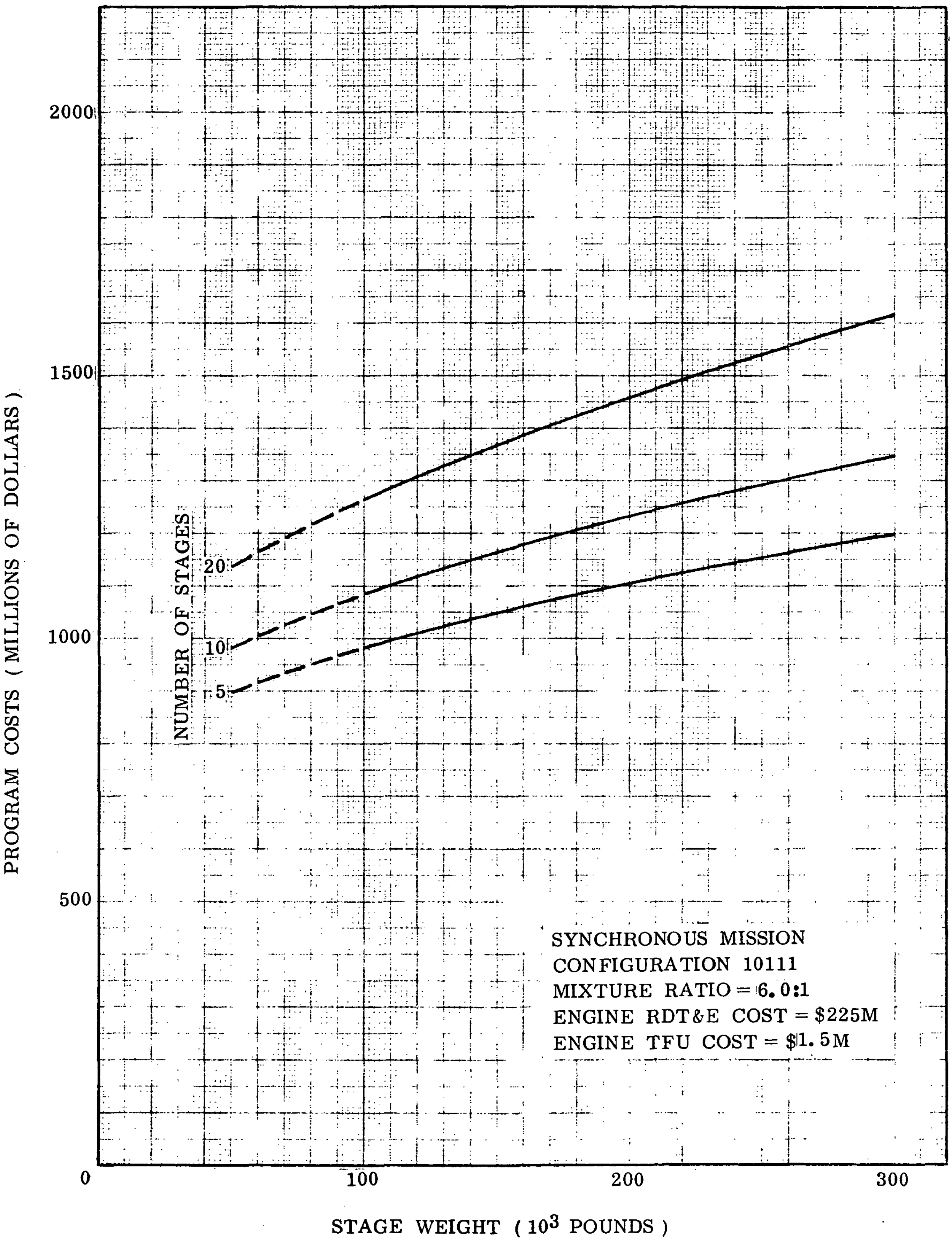

Figure 2-19. Program Costs for a Synchronous Mission (RDT&Eeng=$225M, TFUeng=$1.5M)

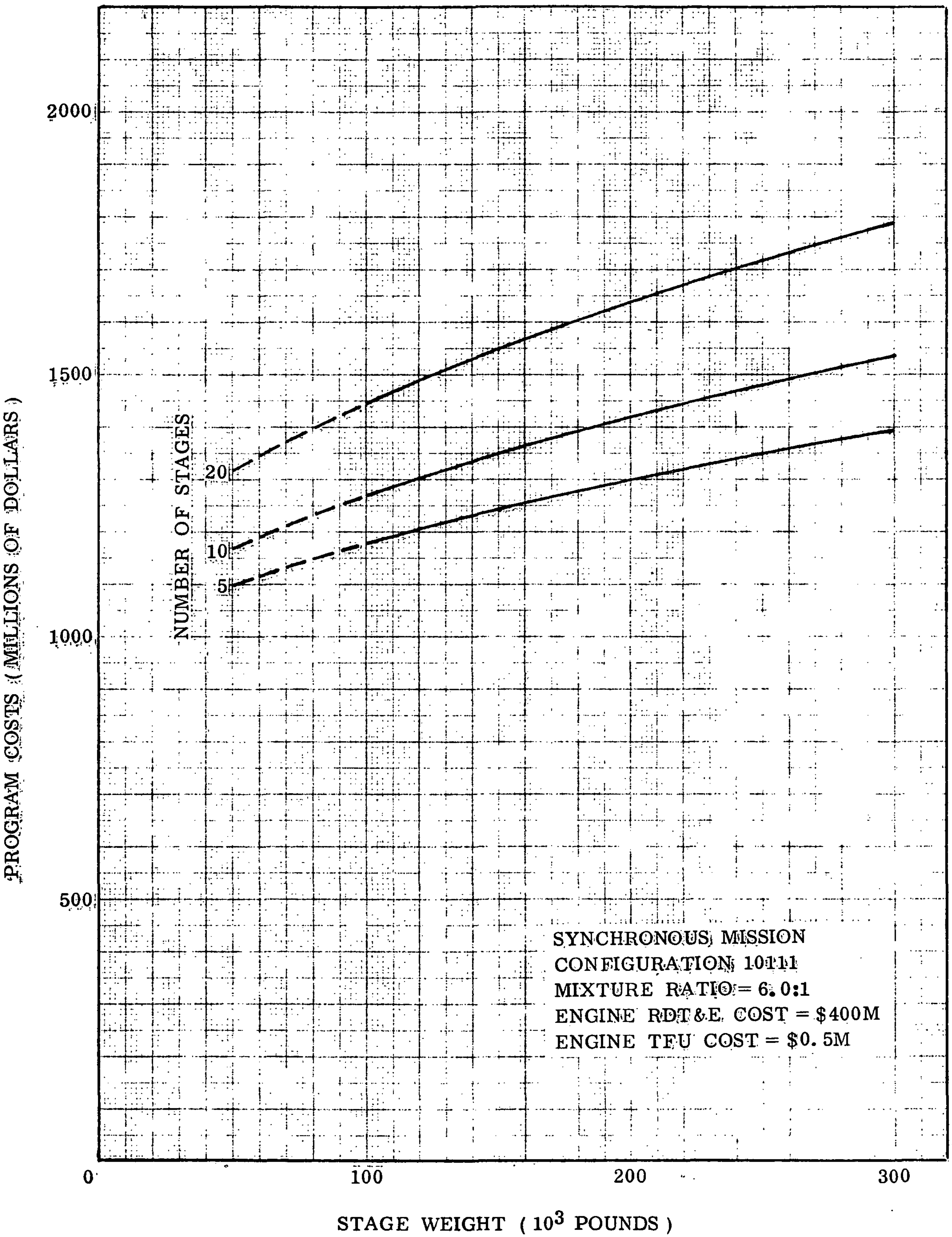

Figure 2-20. Program Costs for a Synchronous Mission (RDT&Eeng=$400M, TFUeng=$0.5M)

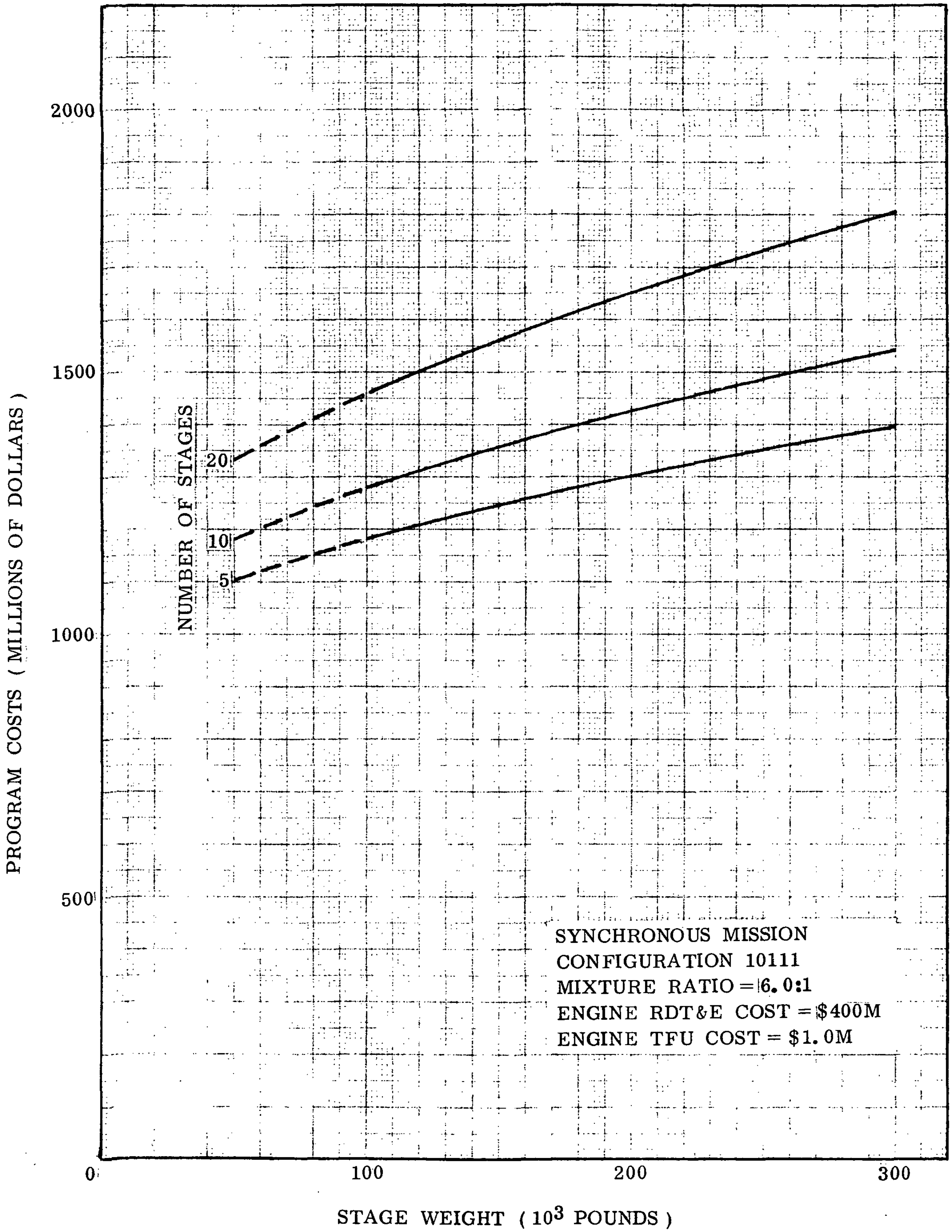

Figure 2-21. Program Costs for a Synchronous Mission (RDT&Eeng=$400, TFUeng=$1.0M)

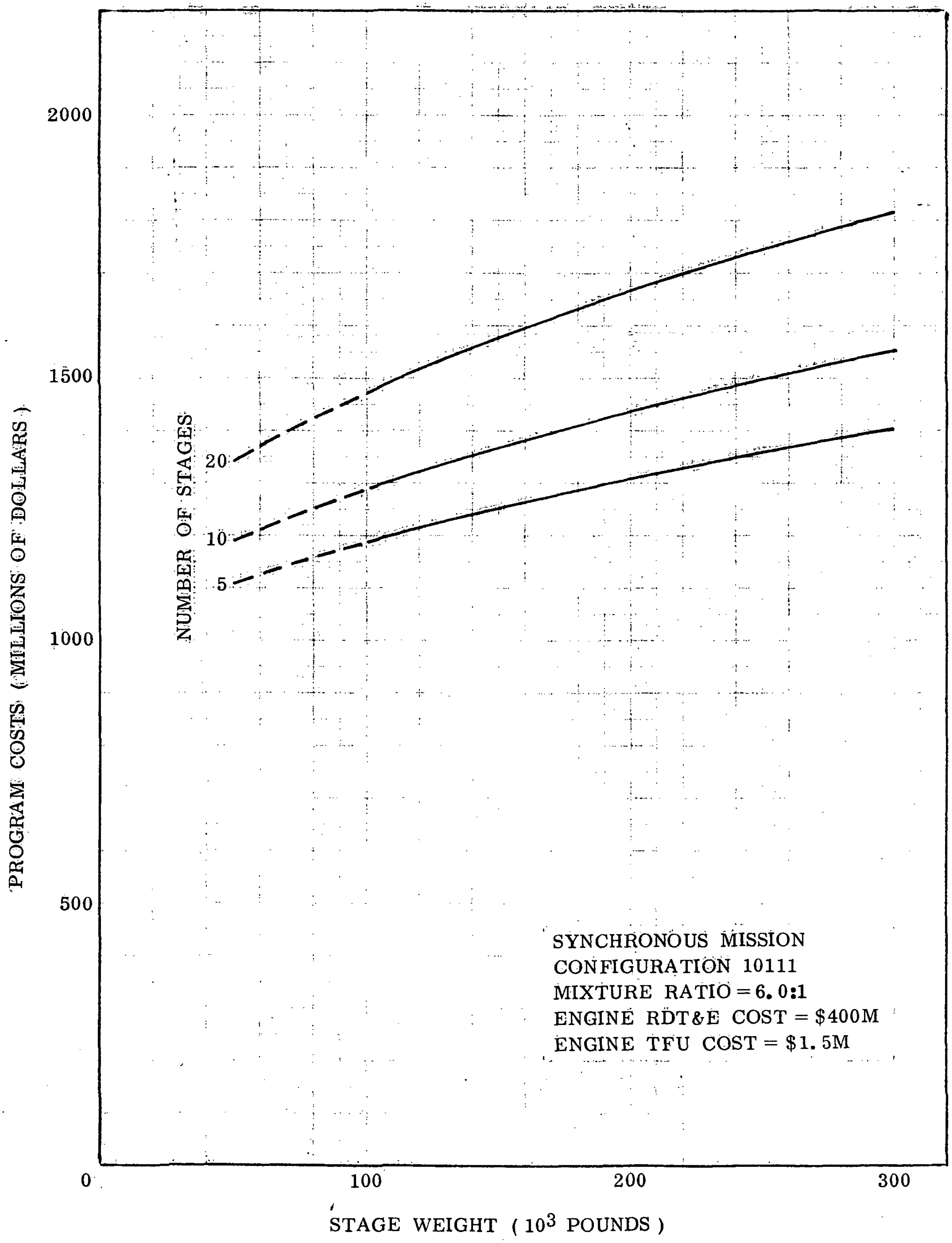

Figure 2-22. Program Cost for a Synchronous Mission (RDT&Eeng=$400M, TFUeng=$1.5M)

relatively insensitive to variations in upper stage weight. Hence, any program cost sensitivities which are determined from these data will be of sufficient accuracy.

The costs of propellants for various sizes of stages are presented in figure 2-23. These costs are based on $0.35/lb and $0.02/lb, for hydrogen and oxygen, respectively. They have been included so that missions which required reusable stages could be priced.

2.2.3 Planetary Missions

2.2.3.1 Mission Profiles

In addition to the data developed for the synchronous mission, general LOX/hydrogen stage data were generated for two planetary missions. The first corresponded to a mission in which a relatively small stage circularlized a payload into an orbit about the planet Mars after a long interplanetary coast. The stage was assumed to be placed on a Mars trajectory by another stage or booster. This mission profile, for this single-burn case, is shown in figure 2-24. The selected coast times and ΔV's correspond roughly to a typical Mars mission.

The second planetary mission required a single stage to perform two major burns. The first burn provided the transfer velocity to place the stage on an interplanetary Mars trajectory. The second burn performed the same function as discussed for the previous planetary mission; i.e., circularization into an orbit about Mars. The mission profile for the two-burn Mars stage is shown in figure 2-25. Again, velocities and coast times were selected to approximate a typical Mars mission.

Hereafter, the two cases are referred to as the single-burn and dual-burn Mars missions.

2.2.3.2 Stage Weights for the Mars Missions

Stages capable of carrying payloads ranging from 0 to 10,000 pounds on the two selected Mars missions were sized for engine specific impulses of 445, 465 and 475 seconds, and engine weights of 100, 1000 and 3500 pounds. As with the synchronous mission, these values of specific impulse and engine weight were selected to ensure that most possible combinations were included in the analysis. The results of the interplanetary stage sizing are presented in figures 2-26 through 2-28, and in figures 2-29 through 2-31 for the single-and dual-burn Mars missions, respectively.

2.2.3.3 Cost Data for the Planetary Mars Missions

As in the analysis of the synchronous mission, RDT&E, TFU, and program costs were determined in the same manner for the stages sized for the two Mars missions. The stage RDT&E costs, which are independent of engine TFU costs, are given as a function of stage weight in figure 2-32, for the 3 different engine RDT&E costs.

The TFU Mars stage costs are depicted in figure 2-33, for the three different engine TFU costs as a function of stage weight. The TFU stage costs are independent of the engine RDT&E costs.

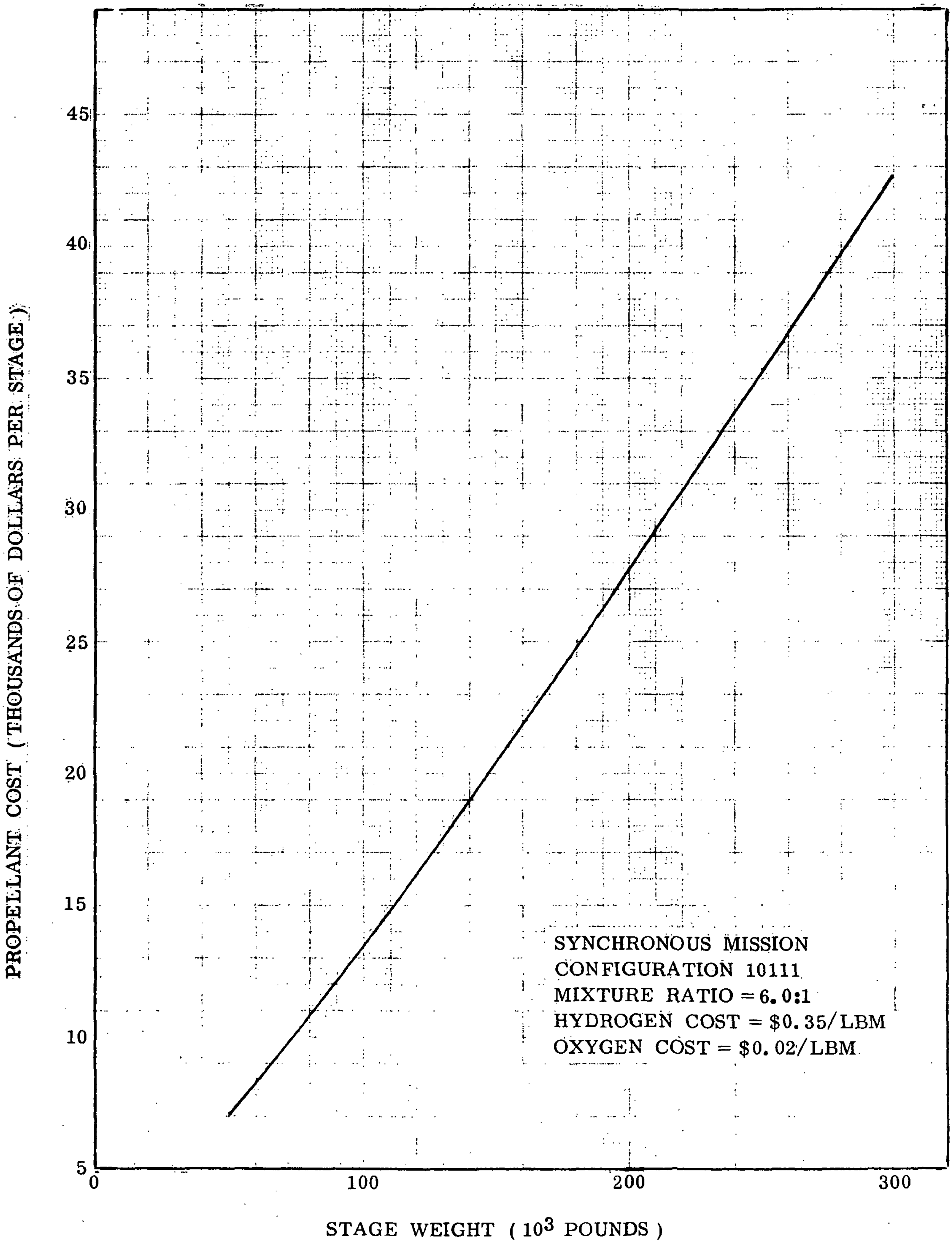

Figure 2-23. Propellant Cost for Synchronous Mission Stages

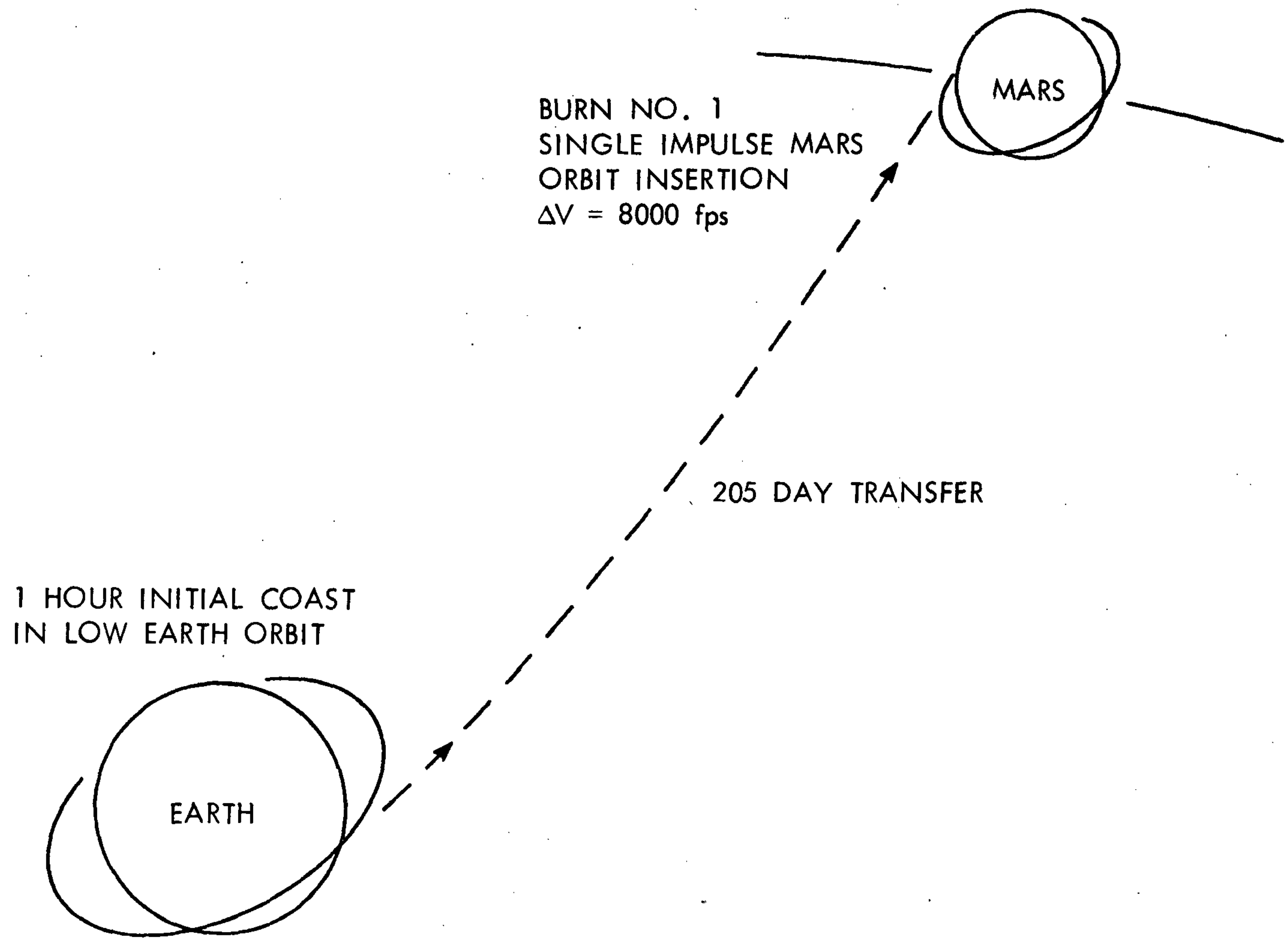

Figure 2-24. The Single-Burn Mars Mission Profile

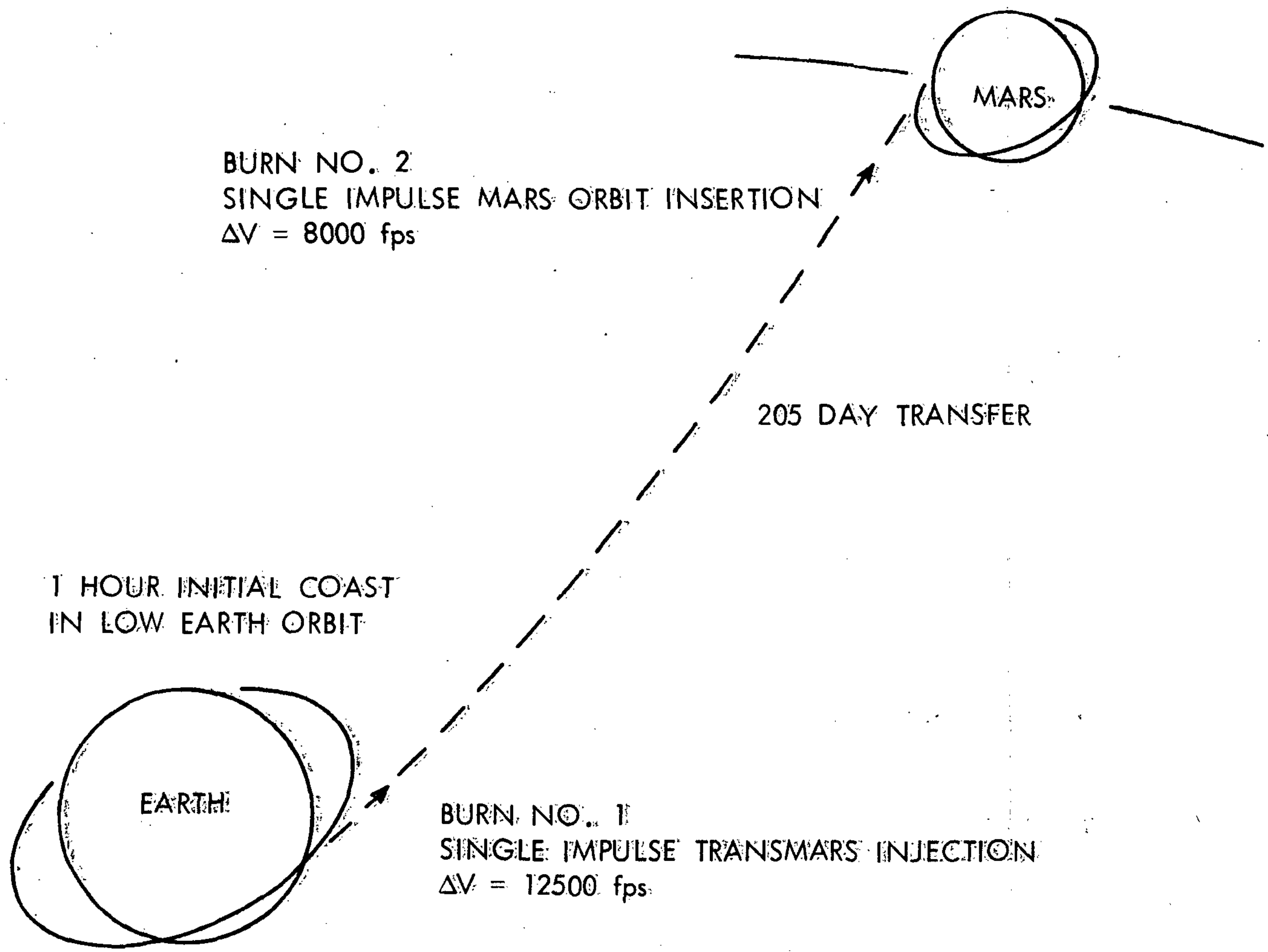

Figure 2-25. The Dual-Burn Mars Mission Profile

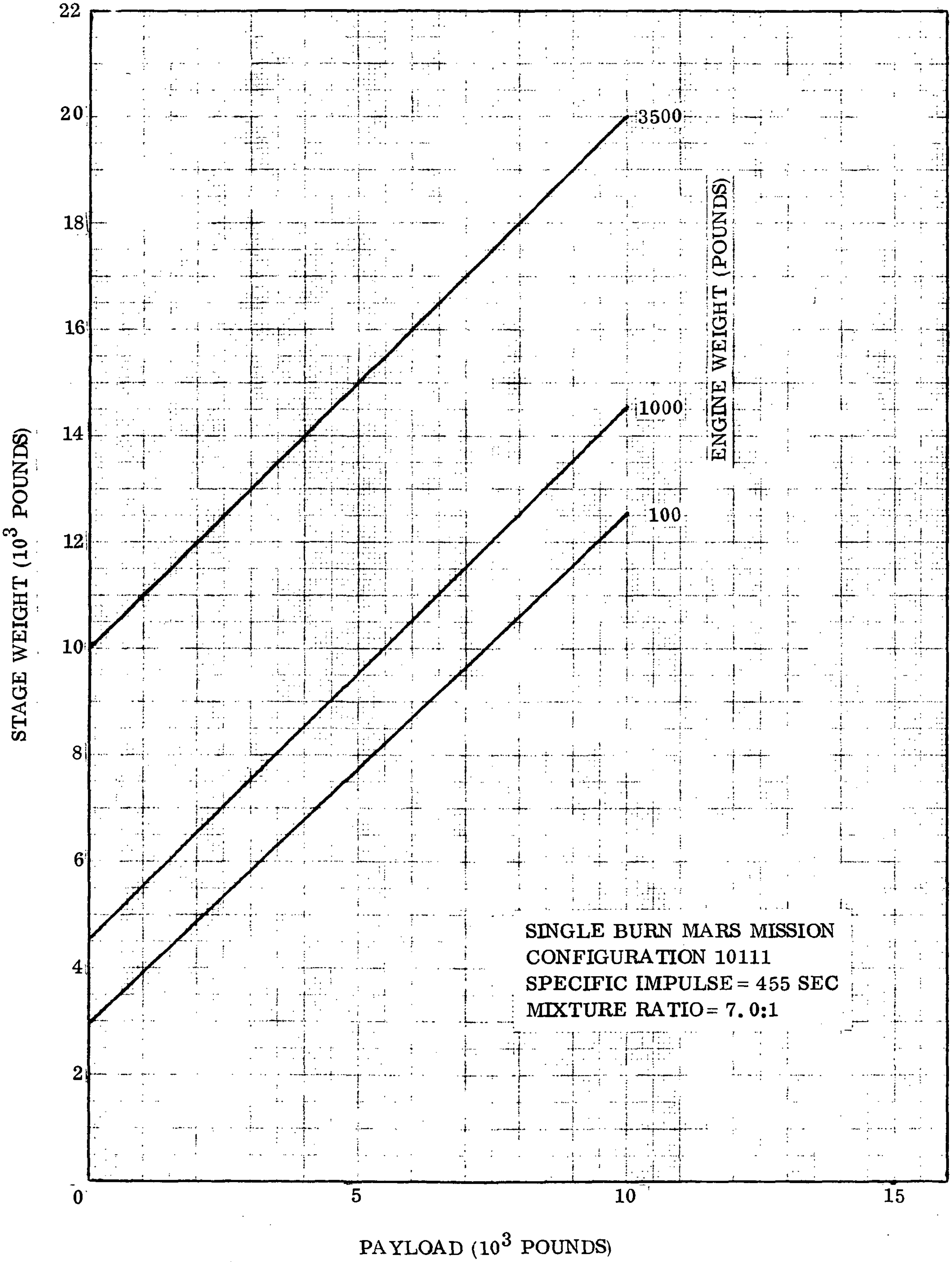

Figure 2-26. Single-Burn Mars Mission Stage Weights (Isp=455)

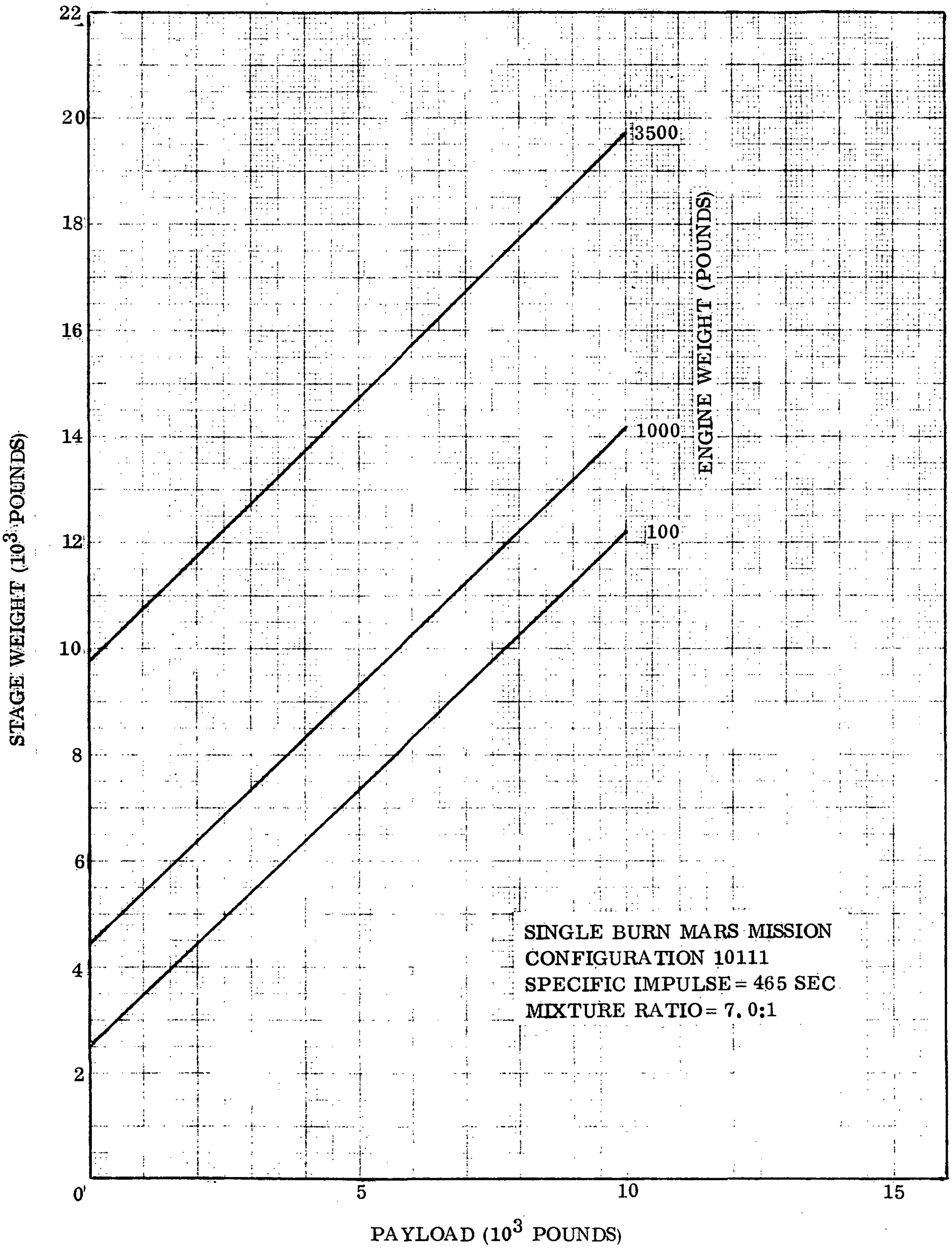

Figure 2-27. Single-Burn Mars Mission Stage Weights (Isp-465)

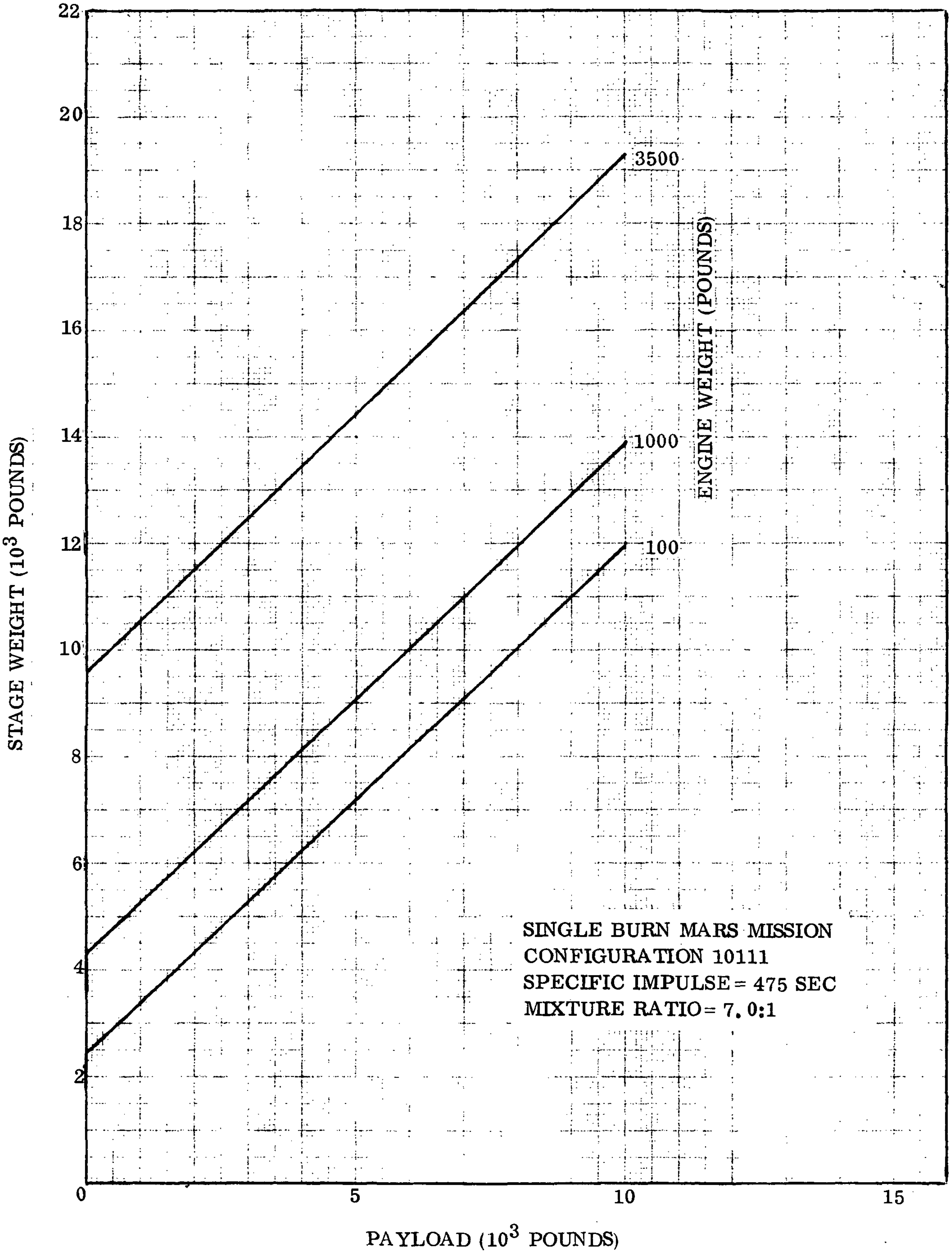

Figure 2-28. Single Burn Mars Mission Stage Weights (Isp=475)

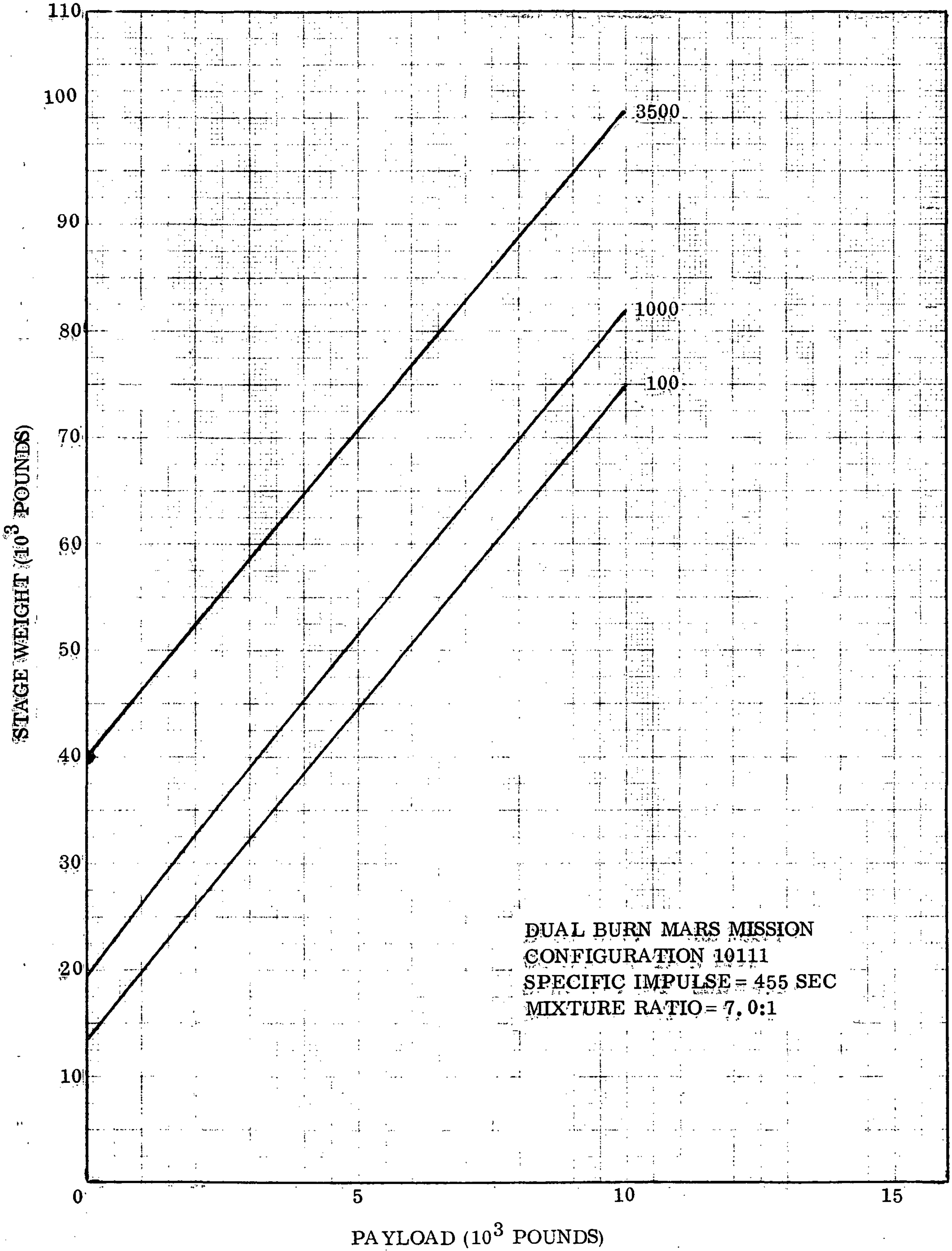

Figure 2-29. Dual-Burn Mars Mission Stage Weights (Isp=455)

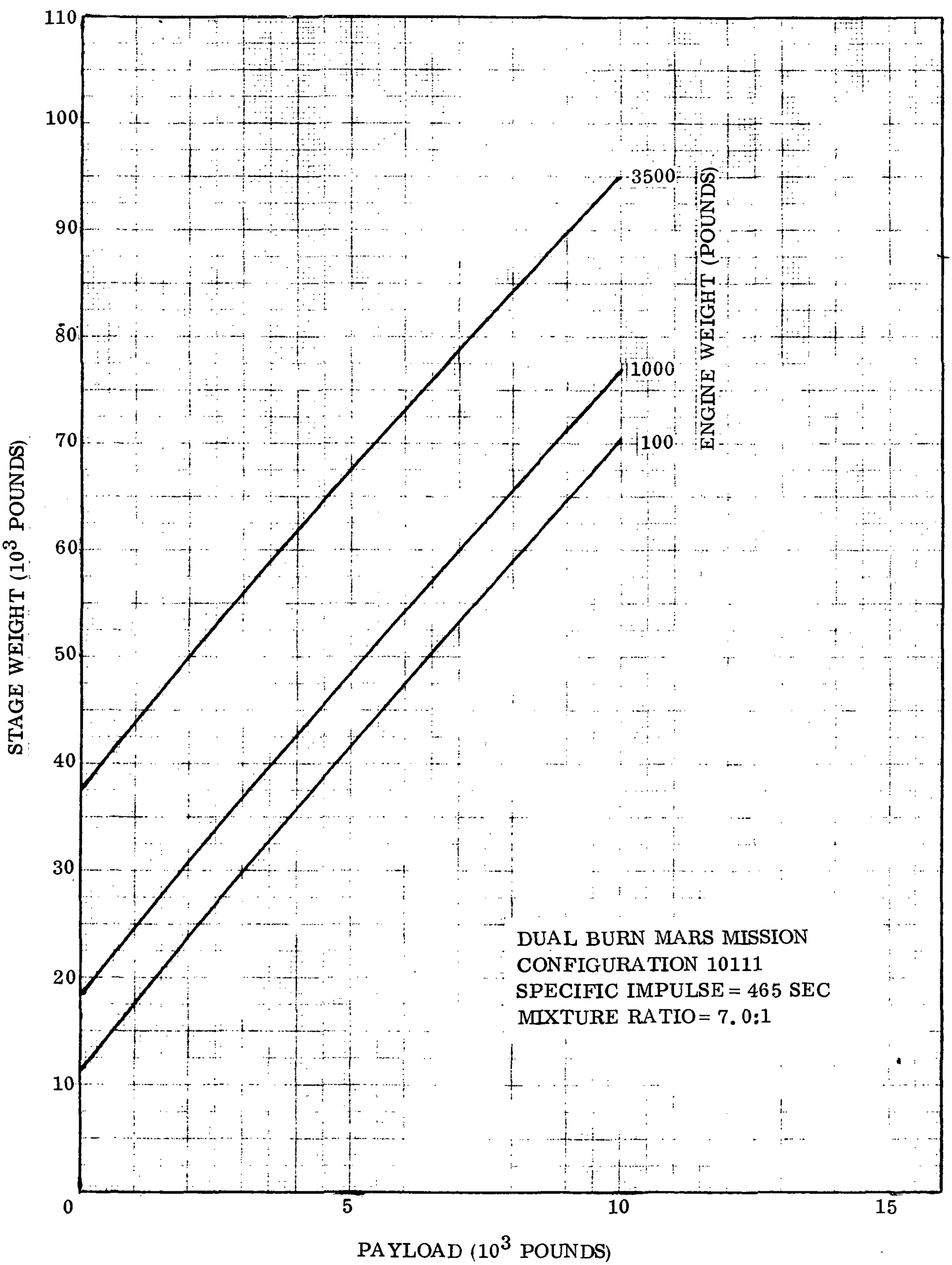

Figure 2-30. Dual-Burn Mars Mission Stage Weights (Isp=465)

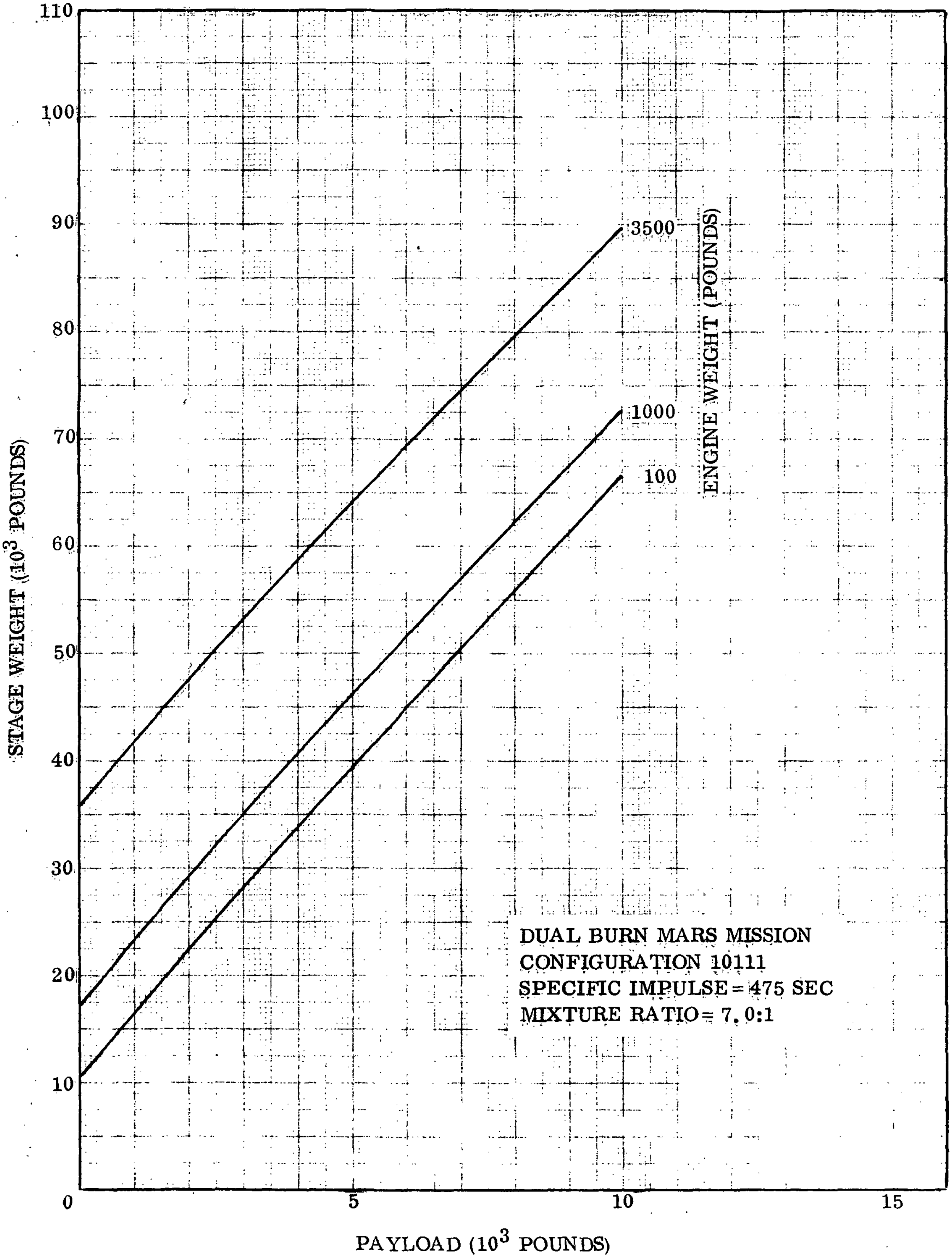

Figure 2-31. Dual-Burn Mars Mission Stage Weights (Isp=475)

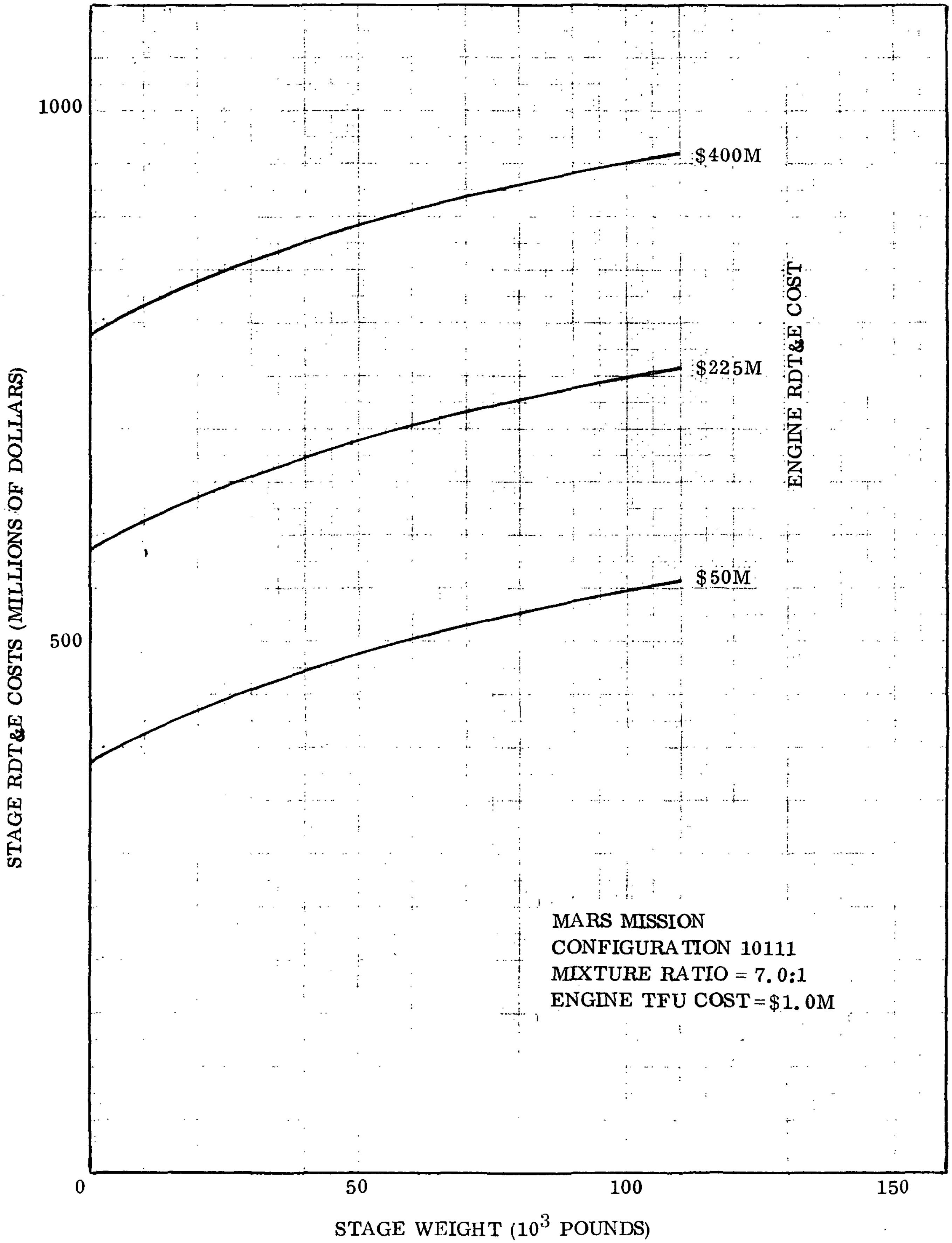

Figure 2-32. Stage RDT&E Costs for a Planetary Mars Mission

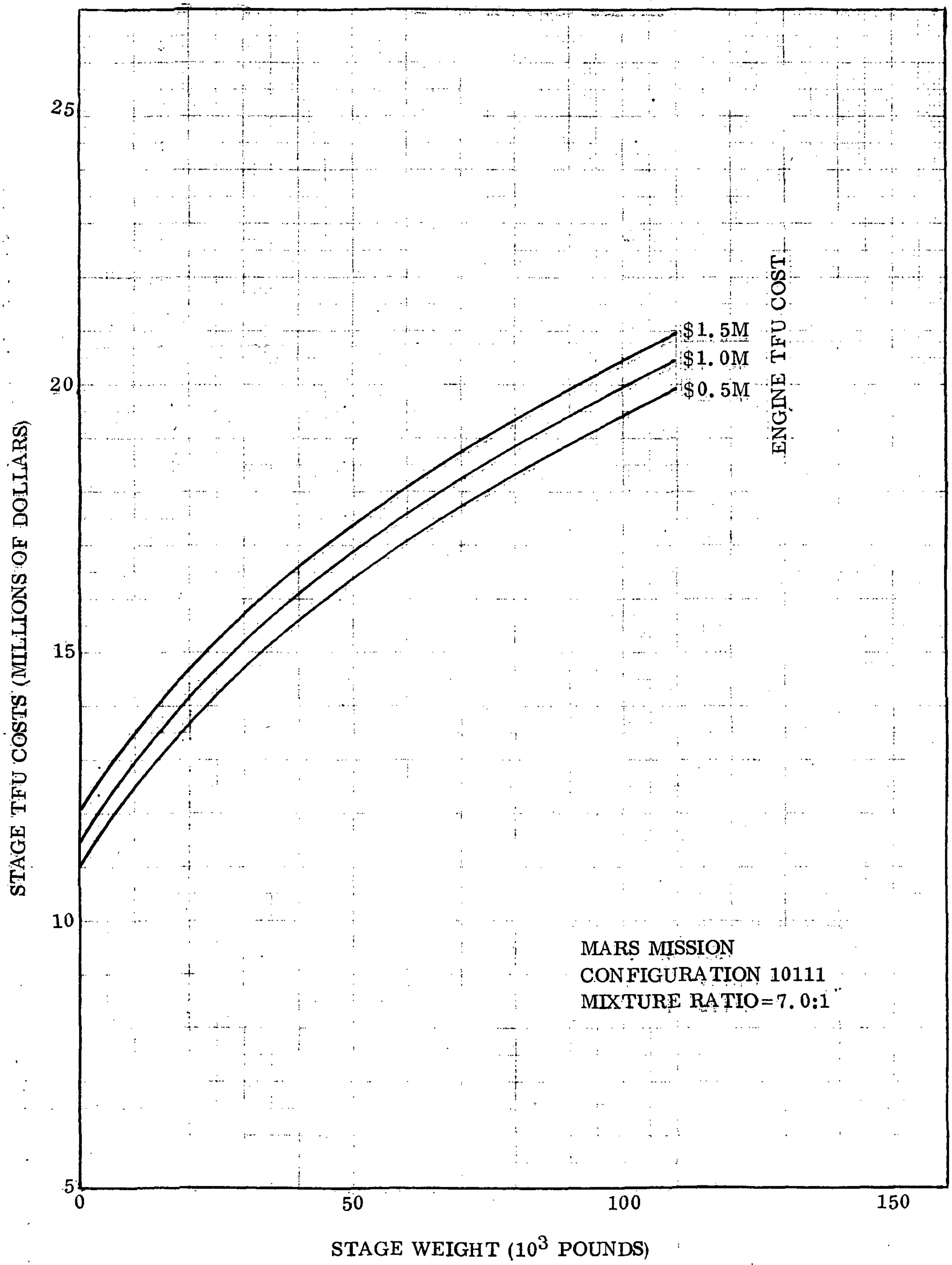

Figure 2-33. Stage First Unit Costs for a Planetary Mars Mission

The program costs for the Mars missions are presented in figures 2-34 through 2-42, for nine combinations of engine RDT&E and TFU costs, as a function of stage weight. As stated above, these costs reflect only the RDT&E, investment, and upper stage propellant costs; they do not contain other cost elements normally associated with operations (see subsection 2.2.1.).

The cost of the upper stage propellants is presented in figure 2-43 for various sizes of stages. These costs are based on $0.35/lb and $0.02/lb for hydrogen and oxygen, respectively.

2.2.4 Sample Solutions

The generalized data presented in the preceding subsections can be used to solve a variety of problems; e.g., the determination of partials of stage weight and cost with respect to specific impulse. In order to reduce the need for interpolation, assume the stage of interest will be designed to retrieve a 15,000-pound payload from synchronous orbit and return it to low earth orbit. Also assume that the engine to be used with the stage has a specific impulse of 465 seconds, a weight of 1000 pounds, and the following costs:

- Engine RDT&E Cost $225M
- Engine TFU Cost $1.0M

The weight of this stage (189,500 lb) can be determined from figure 2-7. The stage weights for the same mission and engine weight, but with specific impulses of 455 and 475 seconds, can be determined from figures 2-4 and 2-10, respectively. These stage weights are shown in table 2-7. Figure 2-44 shows these data in graphic form. The partial of stage weight with respect to specific impulse can be found from the slope of the curve. The weight of a stage corresponding to any specific impulse between 455 and 475 seconds can also be determined from figure 2-44.

Table 2-7. Stage Weights for Various Specific Impulses

I_{sp} (sec)	W_{STAGE} (lb)
455	206,500
465	189,500
475	175,100

Similar data pertaining to the RDT&E, FTU and program costs can be determined from figures 2-12, 2-13 and 2-18, in the same manner. These data are listed in table 2-8, for the three specific impulses, and graphically presented in figures 2-45 through 2-47. The partial derivates, as determined from the figures, are depicted in table 2-9.

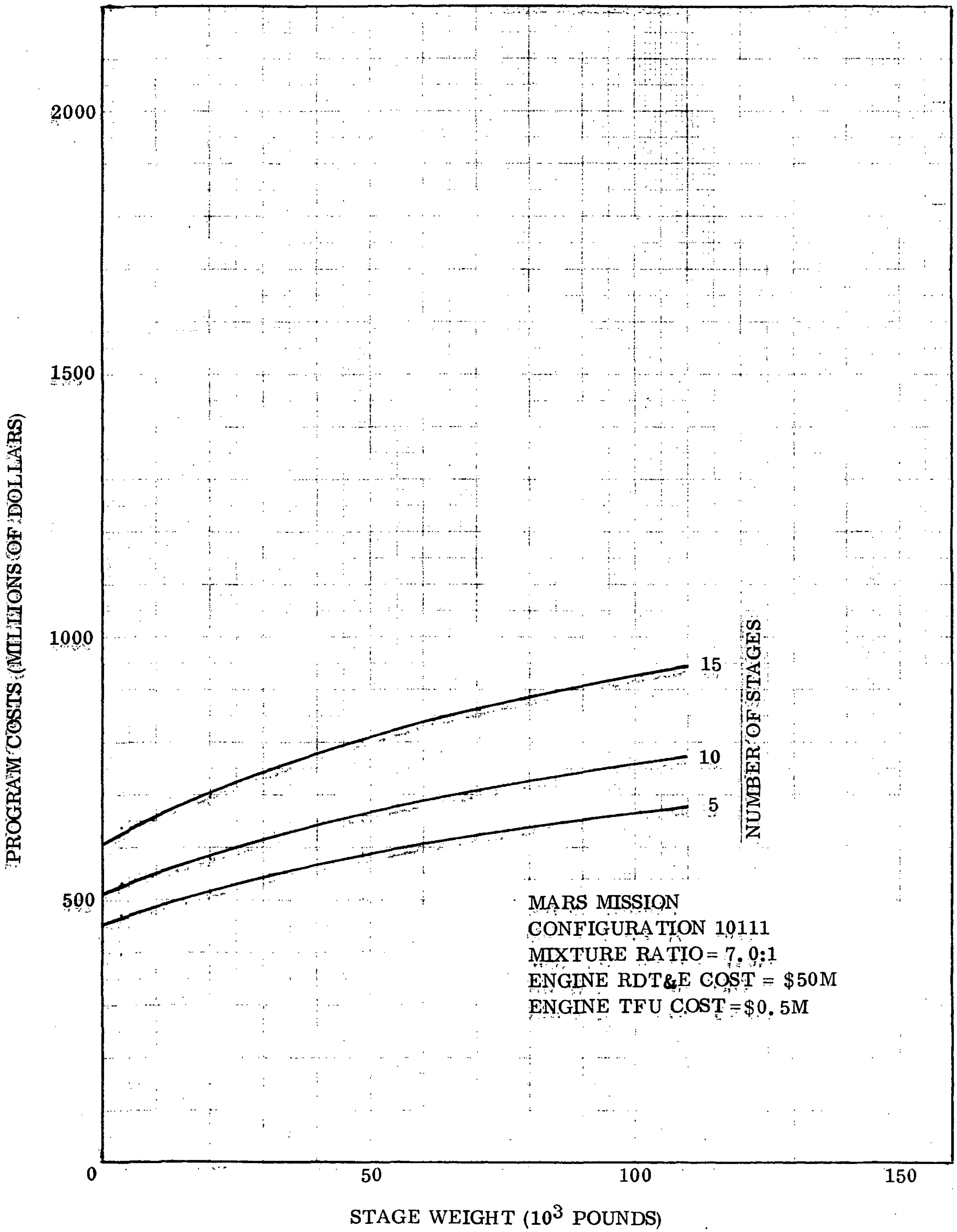

Figure 2-34. Program Costs of a Planetary Mars Mission (RDT&Eeng=$50M, TFUeng=$0.5M)

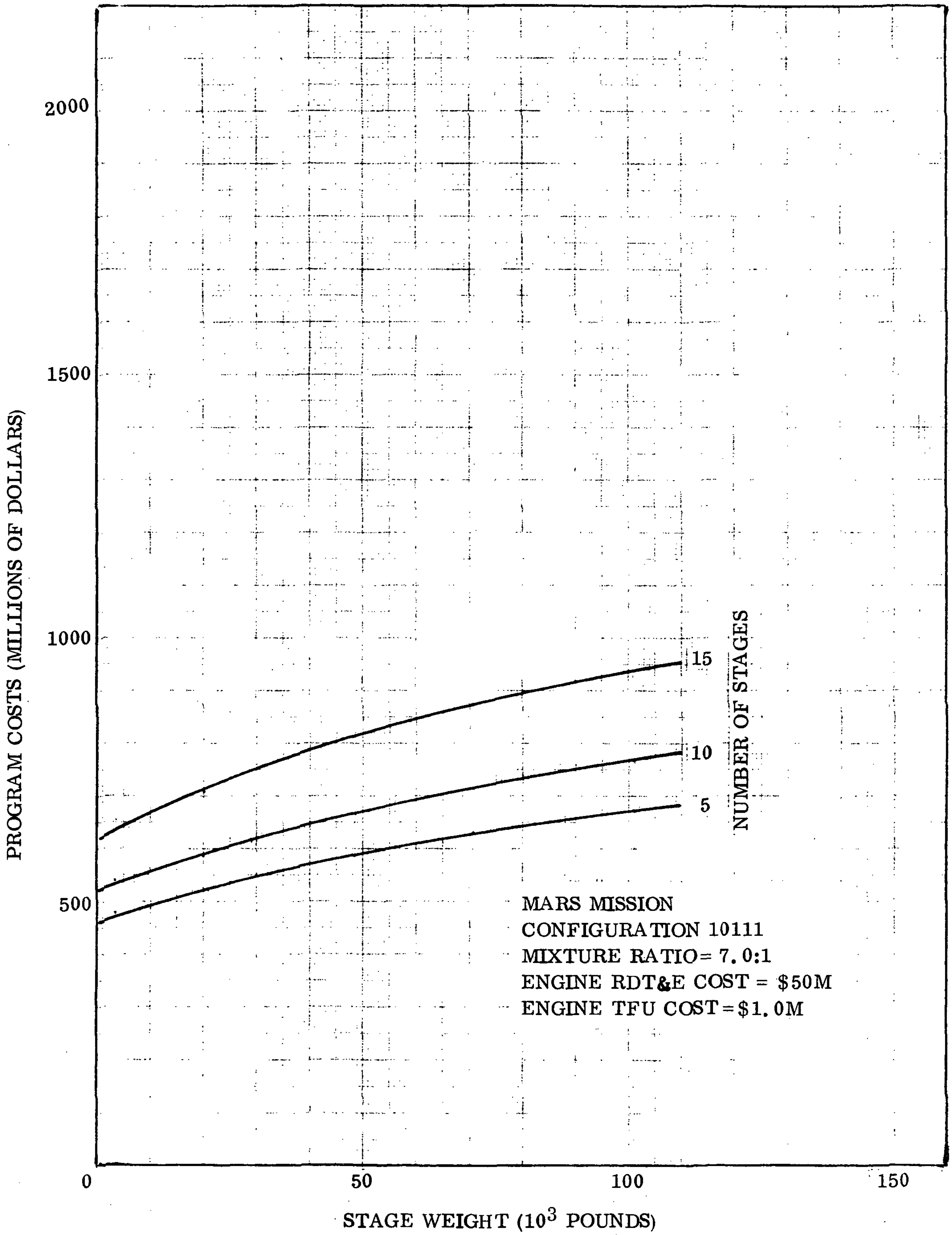

Figure 2-35. Program Cost for a Planetary Mars Missions (RDT&Eeng=$50M, TFUeng=$1.0M)

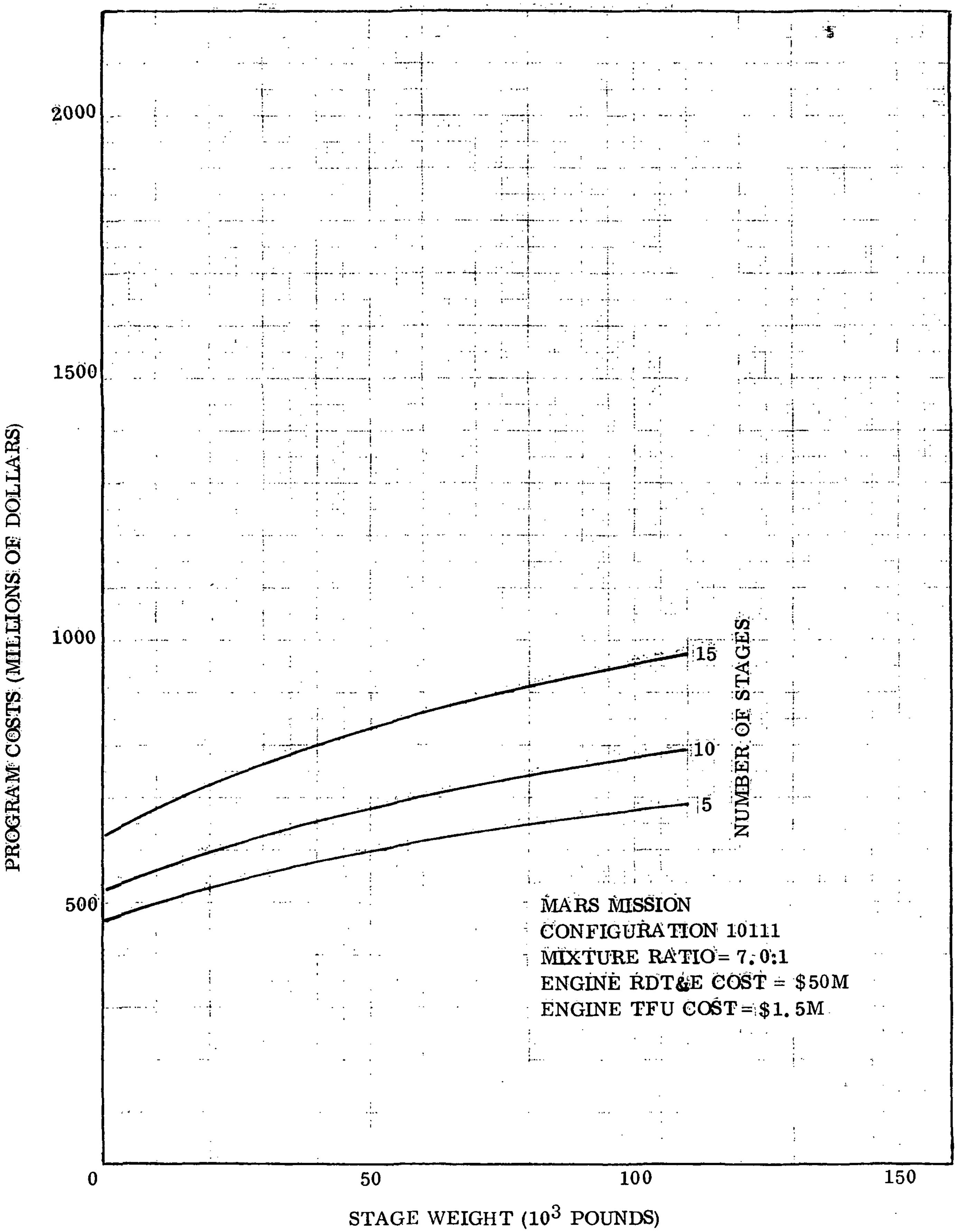

Figure 2-36. Program Costs for a Planetary Mars Mission (RDT&Eeng=$50M, TFUeng=$1.5M)

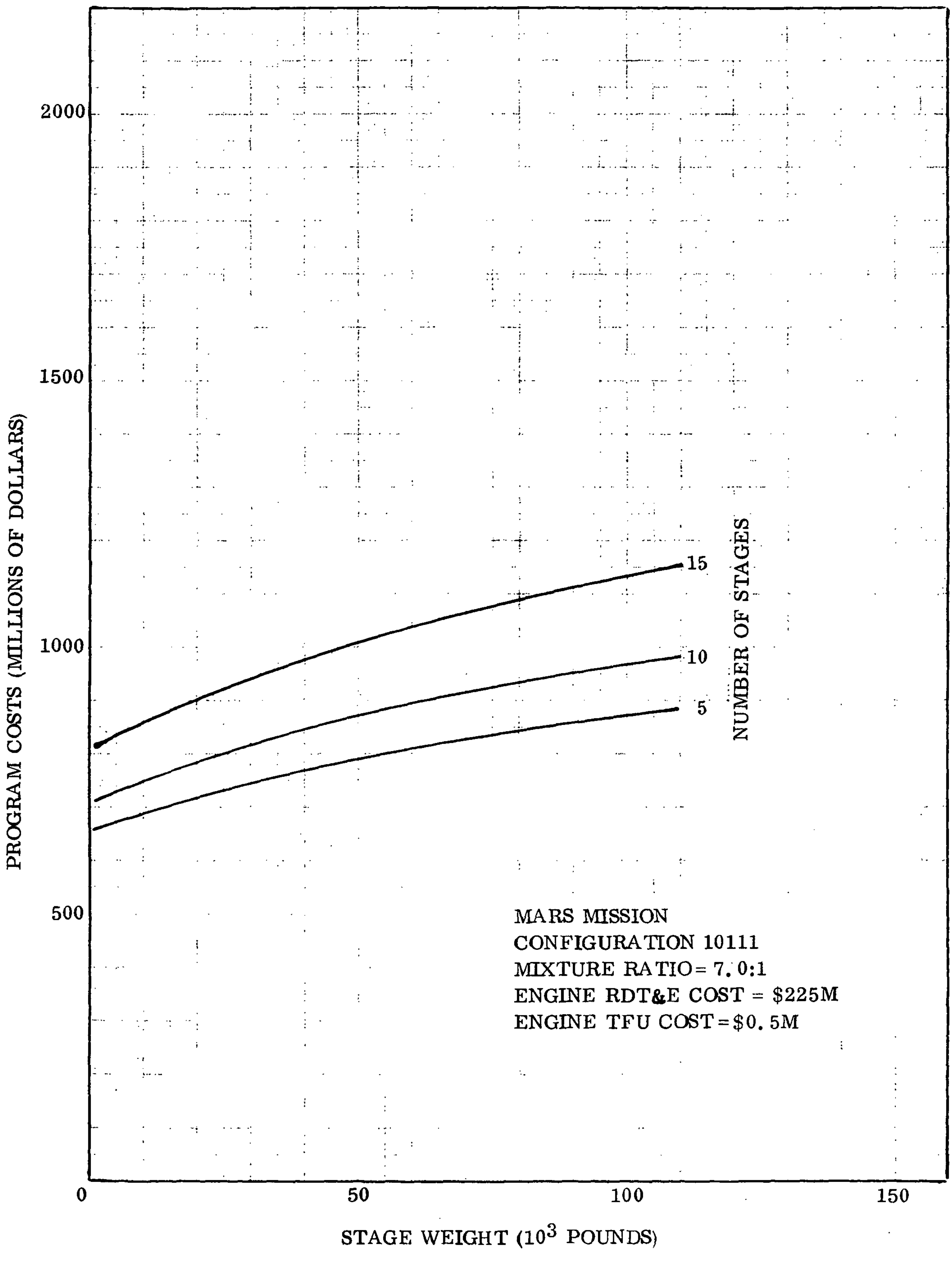

Figure 2-37. Program Costs for a Planetary Mars Mission (RDT&Eeng=$225M, TFUeng=$0.5M)

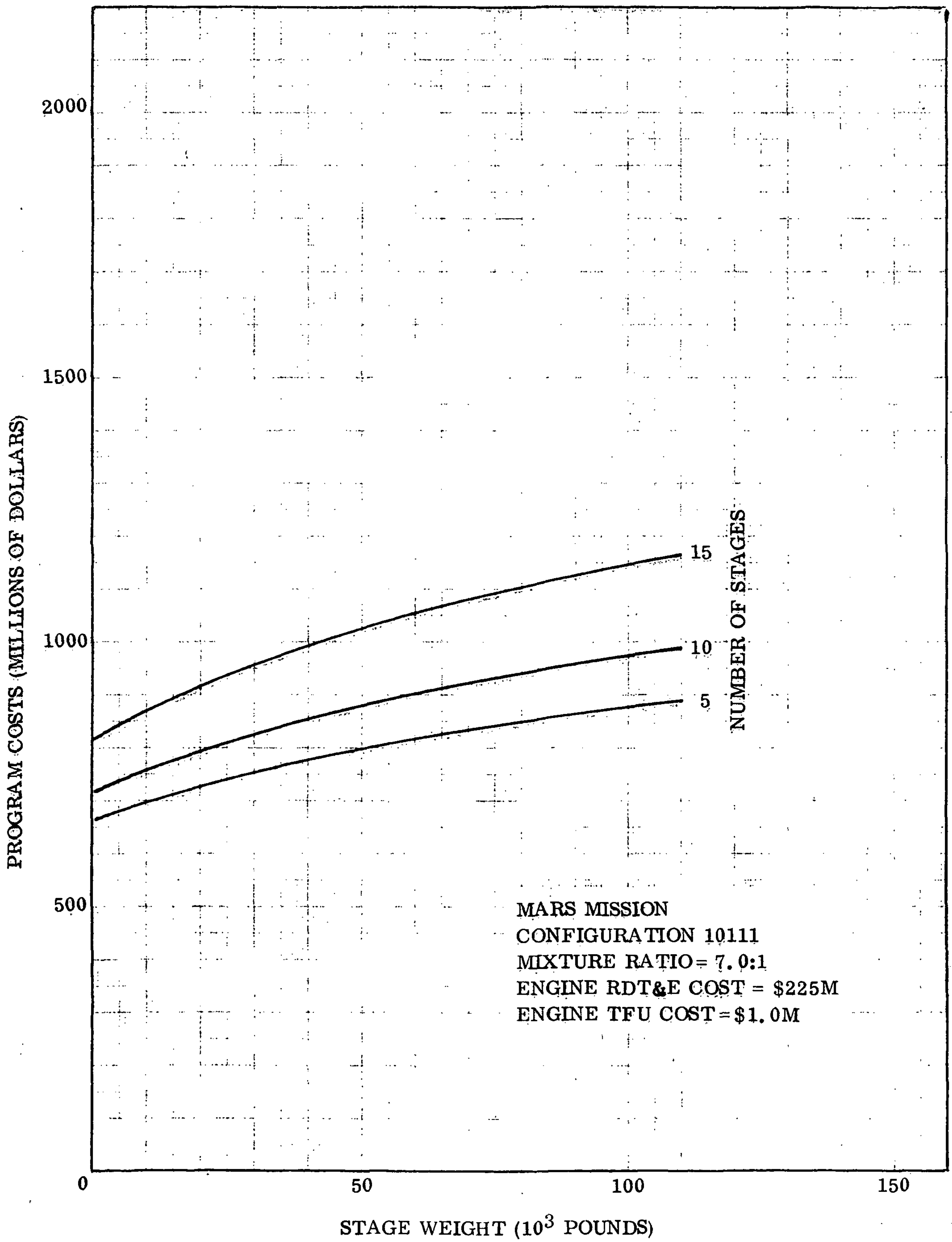

Figure 2-38. Program Costs for a Planetary Mars Mission (RDT&Eeng=$225M, TFUeng=$1.0M)

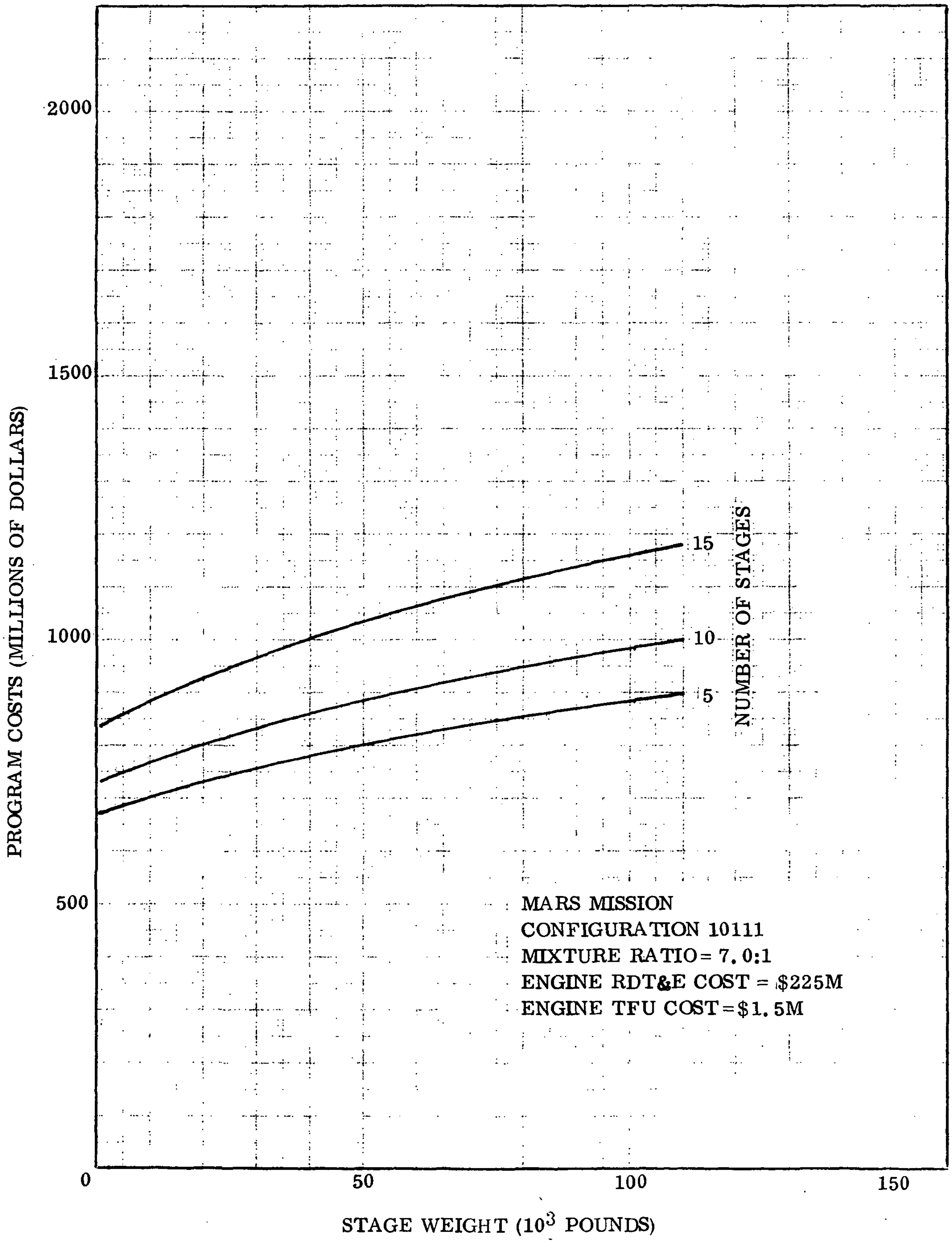

Figure 2-39. Program Costs for a Planetary Mars Mission (RDT&Eeng=$225M, TFUeng=$1.5M)

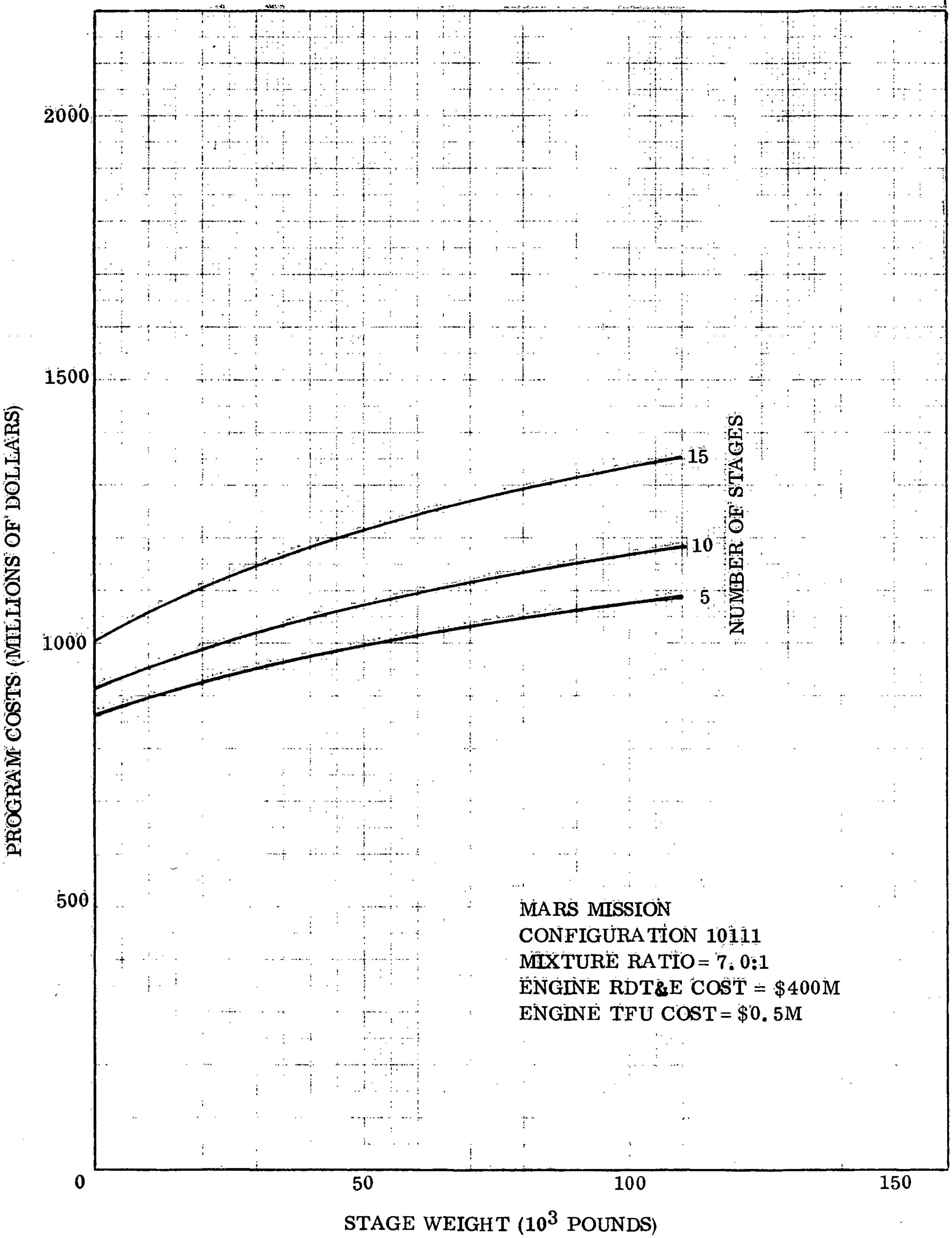

Figure 2-40. Program Costs for a Planetary Mars Mission (RDT&Eeng=$400M, TFUeng=$0.5M)

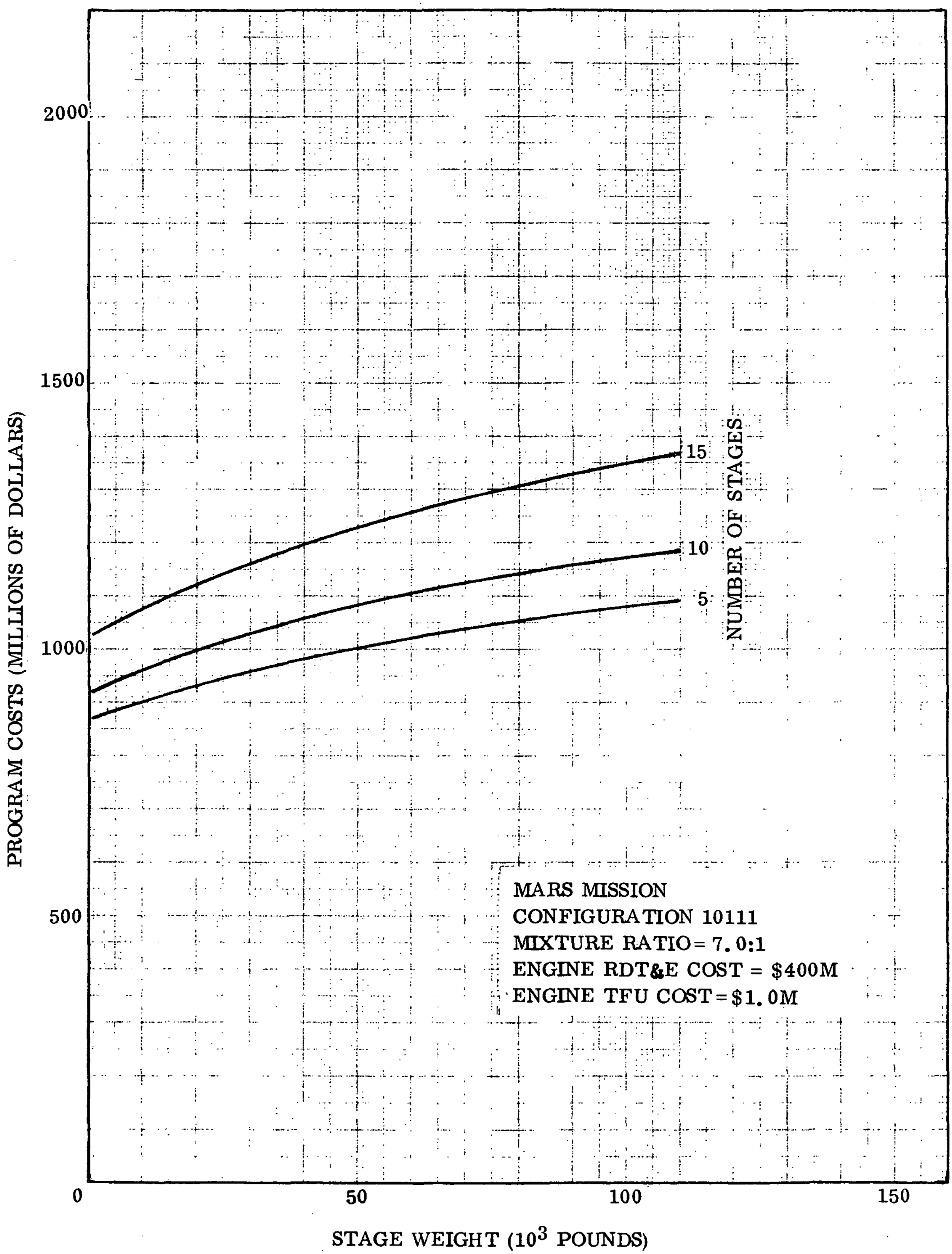

Figure 2-41. Program Costs for a Planetary Mars Mission (RDT&Eeng=$400M, TFUeng=$1.0M)

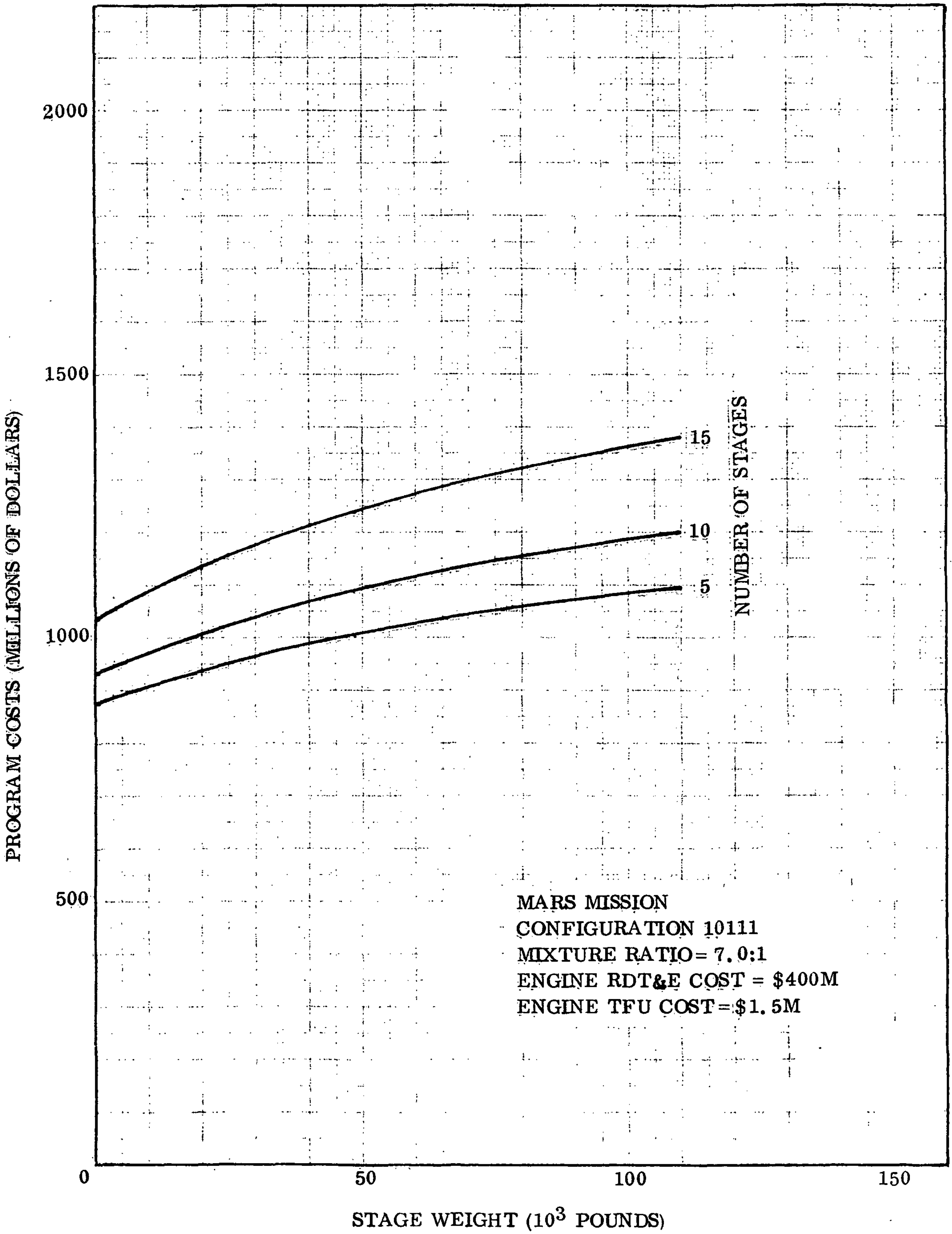

Figure 2-42. Program Costs for a Planetary Mars Mission (RDT&Eeng=$400M, TFUeng=$1.5M)

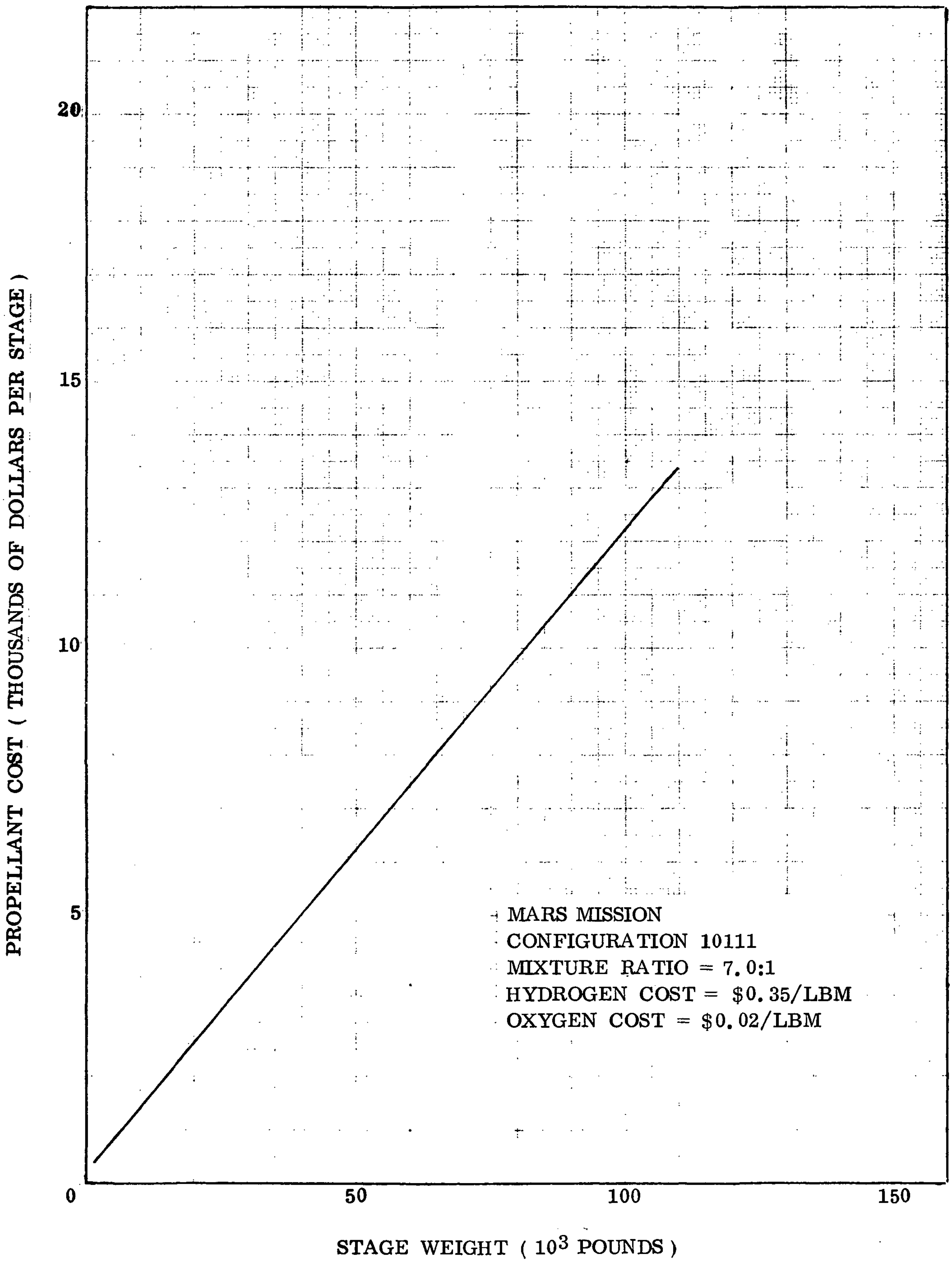

Figure 2-43. Propellant Costs for Planetary Mars Mission Stages

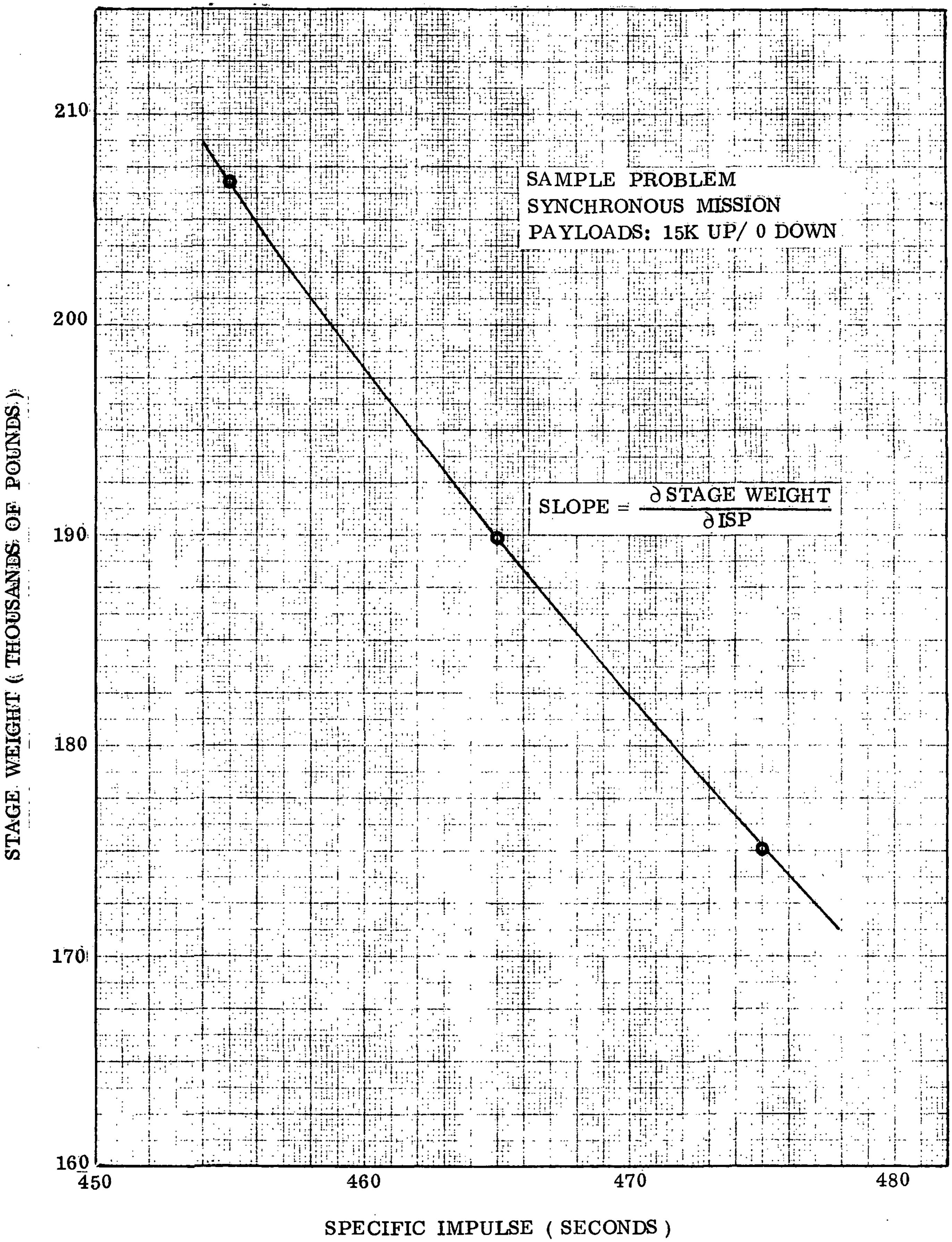

Figure 2-44. Stage Weight as a Function of Specific Impulse

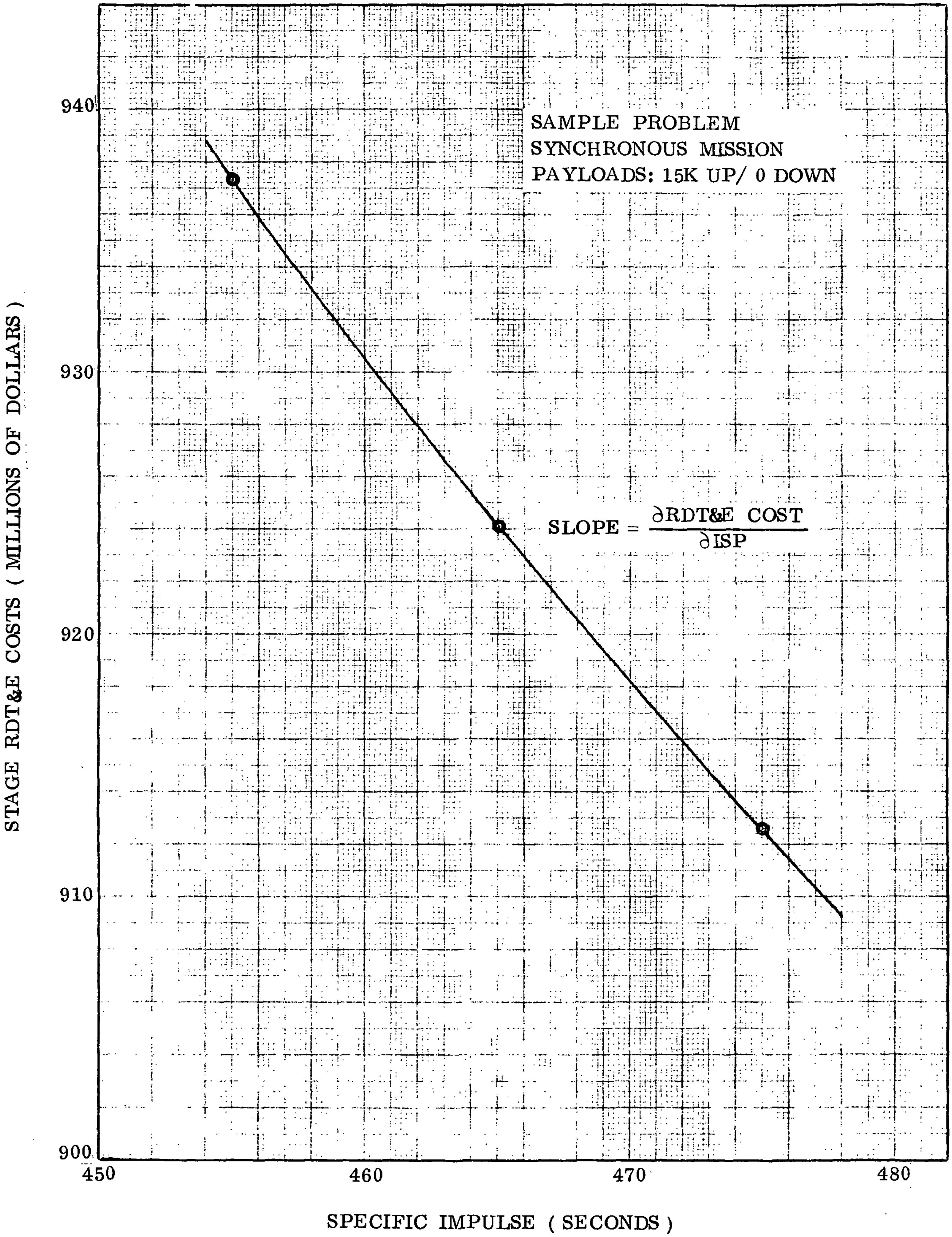

Figure 2-45. Stage RDT&E Costs as a Function of Specific Impulse

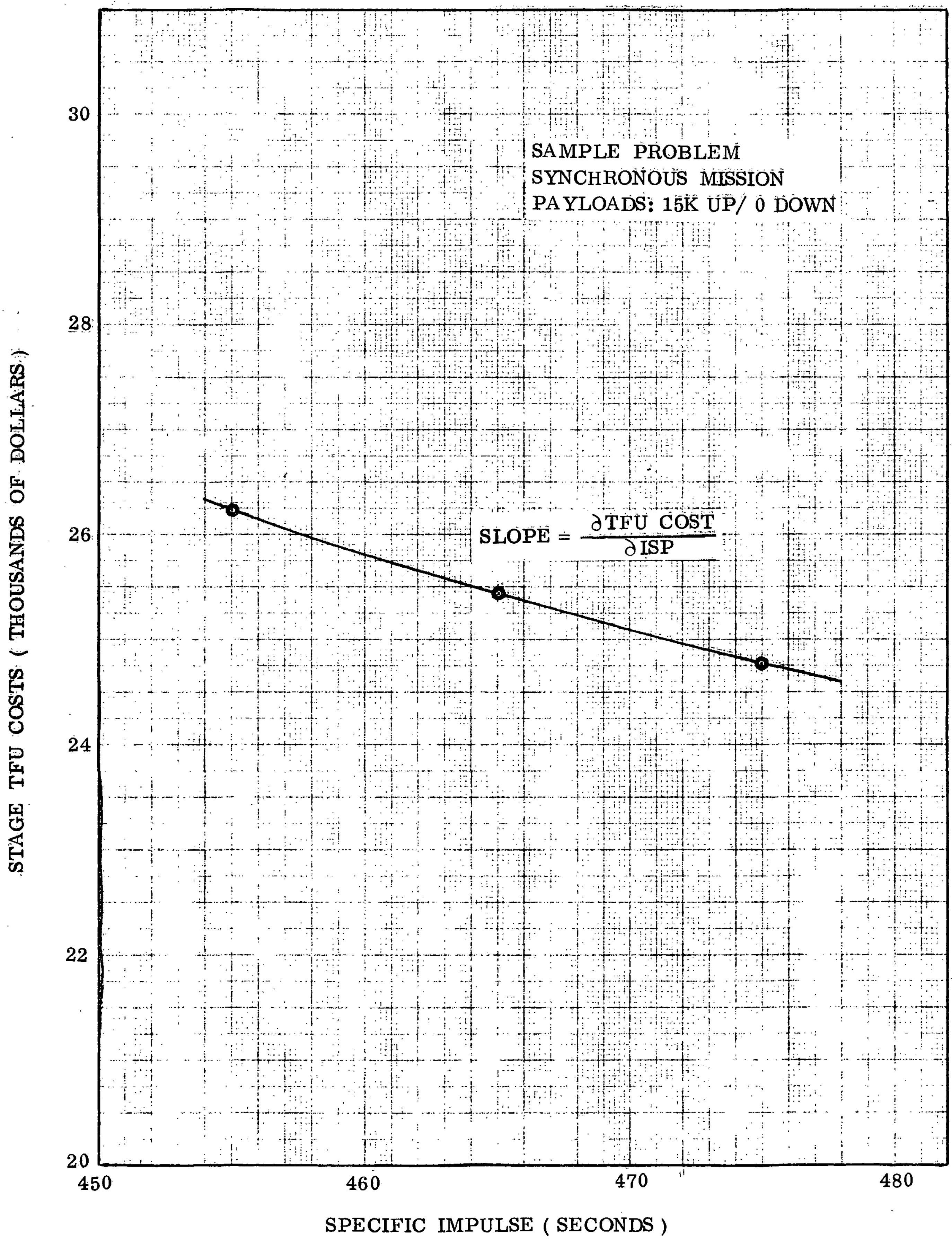

Figure 2-46. Stage First Unit Costs as a Function of Specific Impulse

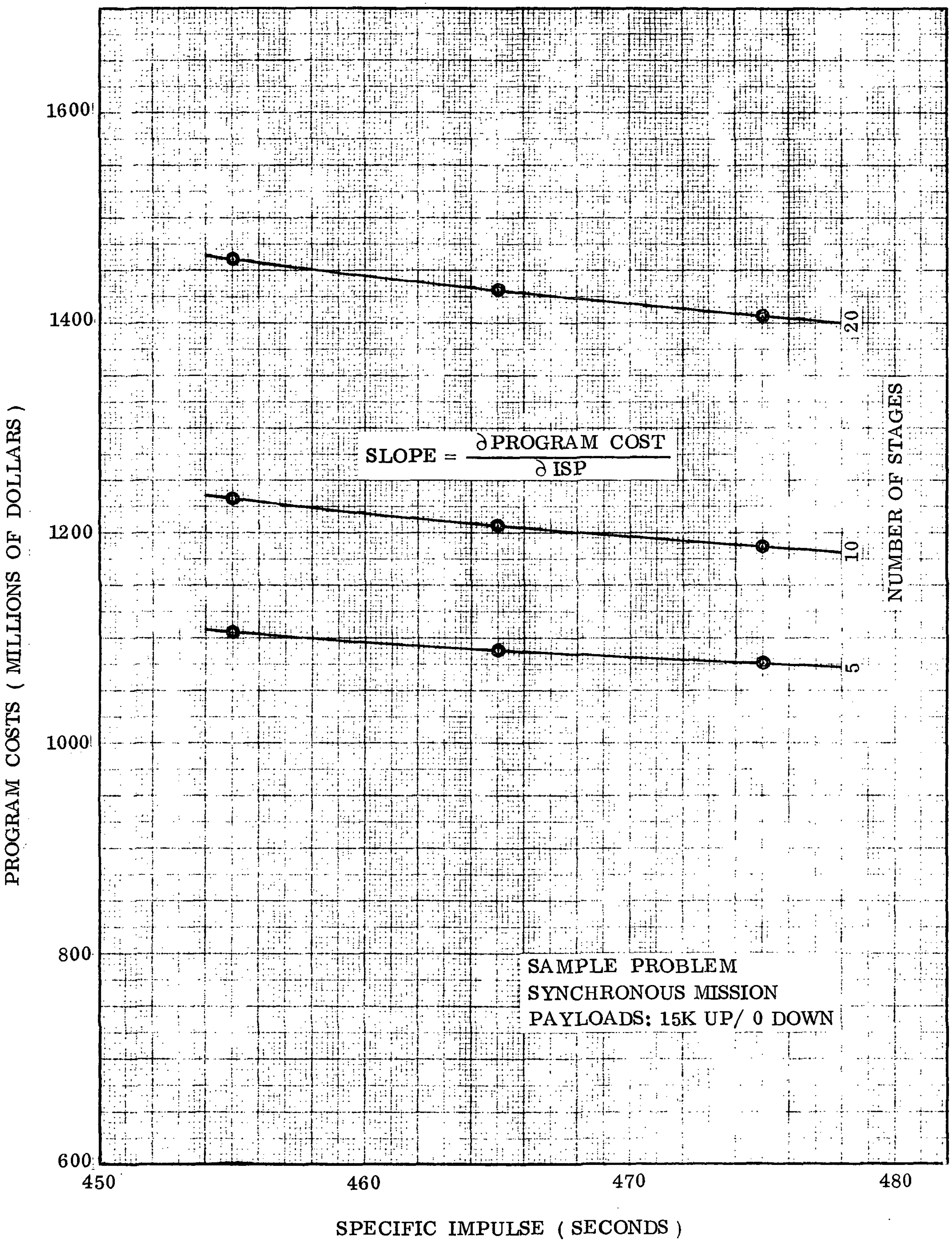

Figure 2-47. Program Costs as Functions of Specific Impulse

Table 2-8. Costs for Various Specific Impulses

I_{sp} (sec)		455	465	475
W_{STAGE} (lb)		206,500	189,000	175,100
RDT&E Cost ($M)		937.3	923.9	912.7
FTU Cost ($M)		26.20	25.43	24.75
Program Cost ($M)	5 Stages	1107.0	1089.0	1075.0
	10 Stages	1231.0	1210.0	1191.0
	20 Stages	1459.0	1431.0	1406.0

Table 2-9. Partials with Respect to Specific Impulse

Stage		$\frac{\partial X}{\partial Isp}$
Stage Weight		-1540 $\frac{lb}{sec}$
Stage RDT&E Cost		-1.23 $\frac{\$M}{sec}$
Stage TFU Cost		-0.0725 $\frac{\$M}{sec}$
Program Costs	5 Stages	-1.60 $\frac{\$M}{sec}$
	10 Stages	-2.00 $\frac{\$M}{sec}$
	20 Stages	-2.65 $\frac{\$M}{sec}$

The versatility of the generalized data can be further demonstrated by expanding the problem. Assume that it is desired to increase stage performance by improving the turbo-machinery efficiency through some means which will provide an additional two seconds of specific impulse. The changes in stage weight and cost can easily be computed from the following differential:

$$\Delta X = \frac{\partial X}{\partial I_{sp}} \cdot \Delta I_{sp}$$

That is,

$$\Delta W_{STAGE} = (-1570 \frac{lb}{sec}) \text{ (2 sec)} = -3140 \text{ lb},$$

$$\Delta \text{RDT\&E} = (-1.23 \frac{\$M}{sec}) \text{ (2 sec)} = -\$2.46 \text{ M},$$

$$\Delta \text{TFU} = (-0.0725 \frac{\$M}{sec}) \text{ (2 sec)} = -\$0.145\text{M},$$

$$\Delta \text{PROGRAM}_{5} = (-1.60 \frac{\$M}{sec}) \text{ (2 sec)} = -\$3.20 \text{ M},$$

$$\Delta \text{PROGRAM}_{10} = (-2.00 \frac{\$M}{sec}) \text{ (2 sec)} = -\$4.00 \text{ M},$$

$$\Delta \text{PROGRAM}_{20} = (-2.65 \frac{\$M}{sec}) \text{ (2 sec)} = -\$5.30 \text{ M}.$$

These delta savings can be utilized to determine the break-even point for the cost associated with the technology improvement. Because the stage being evaluated is larger than the payload capability of the shuttle, the stage must be launched by a regular launch vehicle, and any savings which would be realized by the gain in additional cargo space (due to the smaller stage) need not be taken into consideration. Therefore, the computed program savings will be equal to the break-even point in the cost of technology improvement. The number of stages which must be built in order to break-even can be determined by plotting the computed program saving data versus the number of stages in the program. The data are depicted graphically in figure 2-48, which shows the number of stages that must be built in order for the cost associated with the improvement in technology to equal the savings obtainable in the program. If a larger number of stages is built than is indicated for a given technology cost, then a net savings would be realized.

2.3 EFFECT OF ULTRA-HIGH CHAMBER-PRESSURE INCREASES

A detailed analysis was conducted to assess the merits of ultra-high chamber-pressure increases. This was accomplished by determining the variation in stage size and cost due to changes in specific impulse and engine weight for chamber pressures ranging from 2500 to 10,000 psi. To ensure that a wide variety of conditions was included in this study, the effect of chamber pressure on both theoretical and delivered specific impulses was investigated for two different missions and a wide range of payloads.

The ultra-high chamber-pressure data developed in this task are presented in subsections 2.3.1 through 2.3.4.

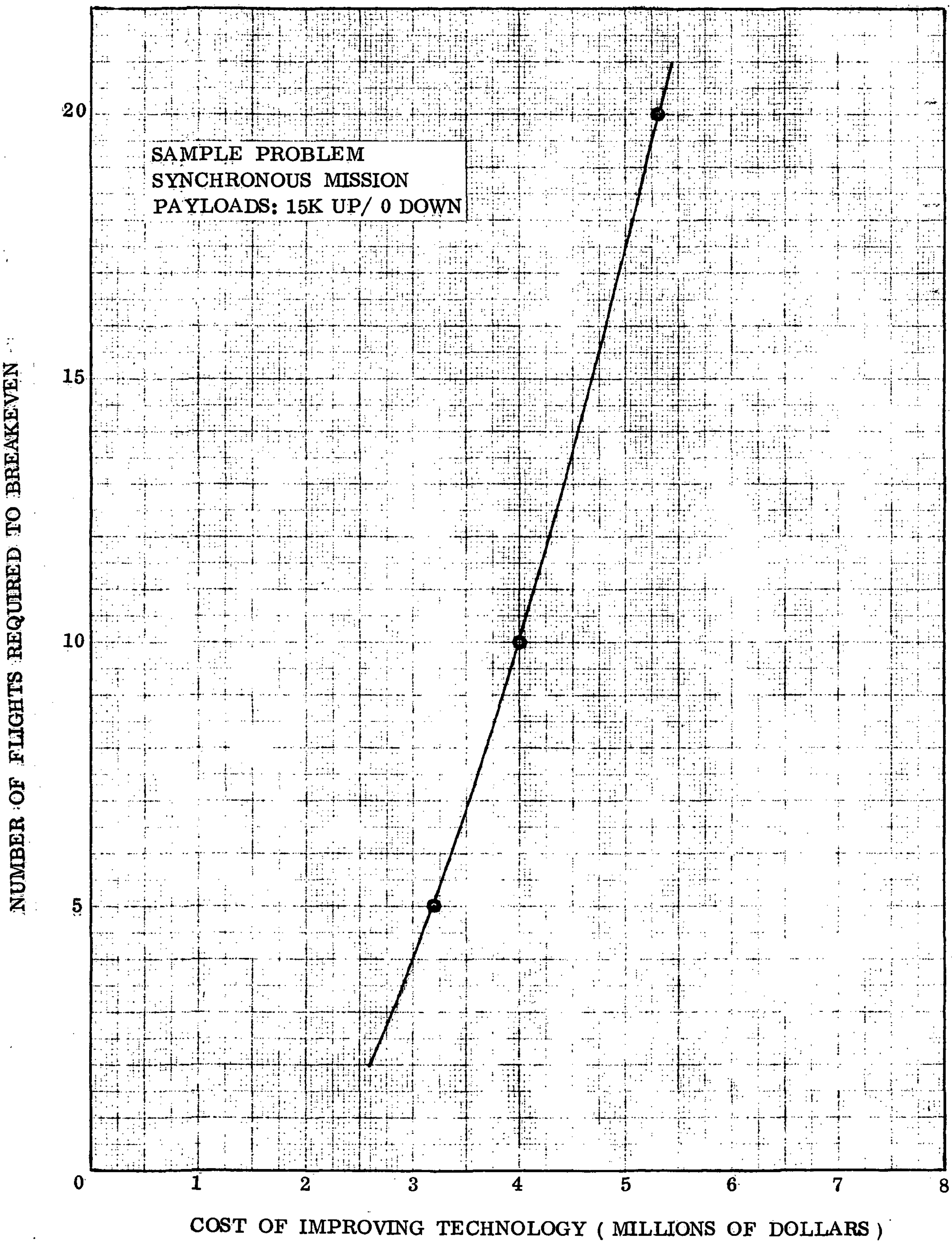

Figure 2-48. Break Even Costs

2.3.1 Data and Assumptions

Throughout the study, certain constraints, guidelines and pertinent design data were used, which are summarized in this subsection. Table 2-10 gives the design constraints used for both the synchronous and Mars missions. Fable 2-11 presents the prime structure data used in computing the weights of the shell and thrust structure. Table 2-12 summarizes the assumed tankage design data including pertinent thermal and meteoroid protection data. These data are shown for each mission. The weights assumed for the astrionics systems and for other miscellaneous systems are given in table 2-13. The subsystem weights shown for the synchronous mission reflect the fail-operational/fail-operational/fail-safe design philosophy currently being considered in the orbit-to-orbit shuttle preliminary designs.

The stage geometry selected as the baseline for this analysis was the tandem tank configuration (10111). This geometry was selected primarily as a matter of convenience because the structure conversion factors (complex-to-monocoque structure weight ratios) used in the sizing program were more accurate for this configuration than for others. A typical stage geometry is illustrated in figure 2-49.

Early in the study, parametric performance data for ultra-high chamber-pressure hydrogen/oxygen engines were obtained from Rocketdyne [5]. These data covered theoretical performance, delivered vacuum specific impulses, vacuum specific impulse efficiency and turbo-machinery efficiency data for both open- and closed-cycle engines. These data cover chamber pressures ranging from 2500 to 10,000 psi. These parametric engine performance data are presented in appendix B.

The engine weights used in this analysis were computed from scaling laws for a fixed-nozzle, pump-fed engine. The equations used for the engine weights were [6]:

$$W_{eng} = 5.0 + 0.0183F + 2.5 \times 10^{-5}\frac{(\epsilon^2 F)}{(Pc)}; \ lb_f \ (1000 \ lb_f \leq F \leq 8000 \ lb_f)$$

$$W_{eng} = 80.0 + 0.0105 F + 2.5 \times 10^{-5}\frac{(\epsilon^2 F)}{(Pc)}; \ lb_f \ (8000 \ lb_f \leq F \leq 8000 \ lb_f)$$

$$W_{eng} = 110.0 + 0.00966F + 2.5 \times 10^{-5}\frac{(\epsilon^2 F)}{(Pc)}; \ lb_f \ (30,000 \ lb_f \leq F \leq 250,000 \ lb_f)$$

where F is the engine thrust in lb_f,
Pc is the chamber pressure in psi, and
ϵ is the nozzle expansion ratio.

The costs generated during this analysis were based on cost estimating relationships which are predicated on historical cost data and the pertinent vehicle parameters [4]. In general, the cost estimating relationships of any cost element contain coefficients, which indicate the technology level and complexity of that individual element. Table 2-14 presents the technology base assumed for the main systems on the stages. Table 2-15 lists the percent learning curves used to compute the investment costs.

A set of reference cost data was developed for the range stage weights considered. The reference RDT&E, TFU, and 20-stage program costs are shown

Table 2-10. Summary of Stage Design Constraints

Constraint \ Mission	Single Stage Synchronous	Interplanetary (Mars)
Maximum Stage Diameter (In.)	260	260
Shell - Tank Spacing (In.)	9.0	9.0
Tank - Tank Spacing (In.)	9.0	9.0
Engine - Tank Spacing Factor (Chamber)	4.0	4.0
Engine - Tank Spacing Factor (Exit)	0.8	0.8
Engine - Booster Spacing (In.)	0.0	0.0
Engine Gimbal Angle (Degrees)	3.0	3.0
Thrust - To - Weight Ratio	0.25	0.25
Axial Acceleration (G's)	1.00	1.00
Lateral Acceleration (G's)	0.05	0.05
Payload Density (Lb/Ft^3)	25.0	25.0
Inert Weight Contingency Factor (%)	7.5	7.5

Table 2-11. Summary of Structural Design Data

Data \ Structure	Shell	Thrust Cone
Material	Aluminum	Aluminum
Density (lb/ft^3)	183.0	183.0
Material Strength (psi)		
Tension	67,000	67,000
Compression	46,000	46,000
Modulus of Elasticity (psi)	10^7	10^7
Safety Factors		
Tension	1.25	1.25
Compression	1.00	1.00
Monocoque-to-Complex Structure Weight Ratio	*	*
Spider Beam Multiplication Factor	N/A	N/A

* A function of diameter and limit load; see appendix C.

Table 2-12. Summary of Tankage Design Data

Data \ Mission	Synchronous	Interplanetary (Mars)
Tankage		
Material	Aluminum	Aluminum
Density (lb/ft^3)	183.0	183.0
Allowable Stress (psi)	60,000	60,000
Factor of Safety	1.10	1.10
Minimum Skin Gauge (In.)	0.025	0.025
Land Factors (Bulkheads)	0.10	0.10
Land Factors (Cylindrical Section)	0.05	0.05
Thermal Protection		
Initial Fuel/Oxidizer Temperature (^oR)	36.0/162.6	36.0/162.6
Initial Fuel/Oxidizer Pressure (psi)	15.0/15.0	15.0/15.0
External Insulation Temperature (^oR)	450.0/470.0	400.0/420.0
Insulation Density (lb/ft^3)	4.5	4.5
Insulation Thermal Conductivity (Btu/Hr-Ft-^oR)	*	*
Meteoroid Protection		
Probability of no Punctures	0.995	0.995
Nominal Mission Altitude (n.m.)	200	500,000
Shield Material	Aluminum	Aluminum
Material Density (lb/ft^3)	183.0	183.0
Material Yield Stress (psi)	70,000	70,000
Minimum Skin Gauges (In.)	0.015	0.015
Miscellaneous		
Minimum Fuel/Oxidizer Ullage Volume (%)	5.0/5.0	5.0/5.0
Residual Fuel/Oxidizer Fraction (%)	2.0/2.0	2.0/2.0
Feedline Flow Velocity (fps)	20.0	20.0
Tank Support Factor	ξ	ξ

* A function of temperature and thickness; see appendix C.

ξ Dependent upon configuration; see appendix C.

Table 2-13. Miscellaneous Subsystem Weights (Pounds)

Subsystem \ Mission	Synchronous	Interplanetary (Mars)
Hydraulic/Pneumatic	550	60
Destruct	50	20
Propellant Utilization	500	50
Communications	400	110
Instrumentation	200	50
Guidance, Navigation and Control	600	250
Electrical	200	30
Electric Power and Distribution	2000	110
Total	4500	680

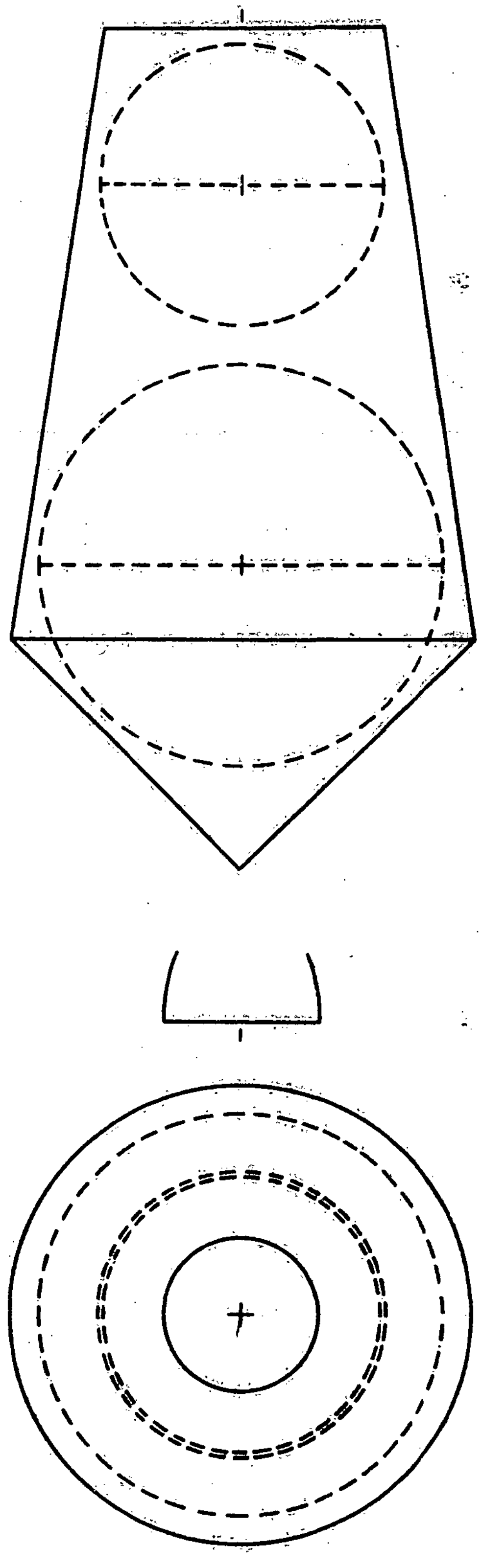

Figure 2-49. Tandem-Tank Stage Configuration (10111)

Table 2-14. Technology Level of Systems

Area	Technology Level & Technique *
Structures	
Shell	SOA - Aluminum Sheet Stringer
Thrust Structure	SOA - Aluminum Sheet Stringer
Tankage	SOA - Aluminum Monocoque
Meteoroid Shield	SOA - Aluminum Monocoque
Tank Supports	ADV - Composite
Propellant Feedlines	SOA - Aluminum
Propulsion	
Main Engines	New, advance, reuseable LH_2/LOX
Reaction Control Thrusters	SOA - Monopropellant
Miscellaneous Subsystems	
Electrical Power and Distribution	Adaptation of existing hardware to a new unmanned, reuseable upper stage
Electrical	
Communication	
Instrumentation	
Guidance, Navigation and Control	
Hydraulic/Pneumatic	
Propellant Utilization	
Destruct	

* SOA - State-of-the-art Technology
ADV - Davance Technology

Table 2-15. Investment Learning Curves

System	Learning Curve
Structures	90%
Propulsion	95%
Miscellaneous Subsystems	90%
Assembly and Checkout	90%

in figures 2-50, 2-51 and 2-52, respectively. These data were used as the basis for determining the cost variations discussed in the subsequent paragraphs of this section.

The program cost data developed during this analysis include only the RDT&E, investment, and upper stage propellant costs. The program costs presented do not contain the other cost elements normally included in the operations costs, because they are mainly launch vehicle and mission model oriented, and for identical missions and similarly sized upper stages, the operational costs are relatively insensitive to variations in upper stage weight. Hence, any program cost sensitivities which are determined from these program cost data will be of sufficient accuracy.

2.3.2 Mission Profiles

The two missions considered in this study were: 1) an earth orbit to synchronous orbit and return, and 2) a two-burn Mars planetary mission. The mission profiles selected for this analysis are discussed in the next two paragraphs.

2.3.2.1 Synchronous Mission Profiles

The profile selected for the synchronous mission was the transfer of payloads between a low-inclination, low-altitude earth (parking) orbit and a synchronous orbit. This mission would require the liquid hydrogen-liquid oxygen stage to transport a payload from low-parking orbit to synchronous orbit and return with the same or a different payload of equal weight.

The typical profile for this mission is depicted in figure 2-53. This involves a Hohmann-type transfer maneuver from low earth orbit to synchronous altitude, a plane change and circularization at synchronous orbit, and a return Hohmann transfer and plane change at synchronous orbit and circularization into the original low earth orbit.

The four velocities used for this mission assumed a Hohmann-type transfer between a 28½-degree inclination, 100-nautical-mile circular orbit and an equational (0-degree inclination) synchronous orbit. The velocities were not corrected to account for the effects of stage initial thrust-to-weight ratio and specific impulse, nor those of orbital regression on the velocity requirements.

2.3.2.2 Mars Mission Profile

In addition to the data developed for the synchronous mission, data were generated for a planetary mission. This mission required a single stage to perform two major burns. The first to provide the necessary velocity to place the stage on an interplanetary trajectory; the second burn to circularize the stage and payload into an orbit about the planet Mars. The mission profile for the two-burn interplanetary mission is shown in figure 2-54. The coast times and velocities were selected to approximate a typical Mars mission.

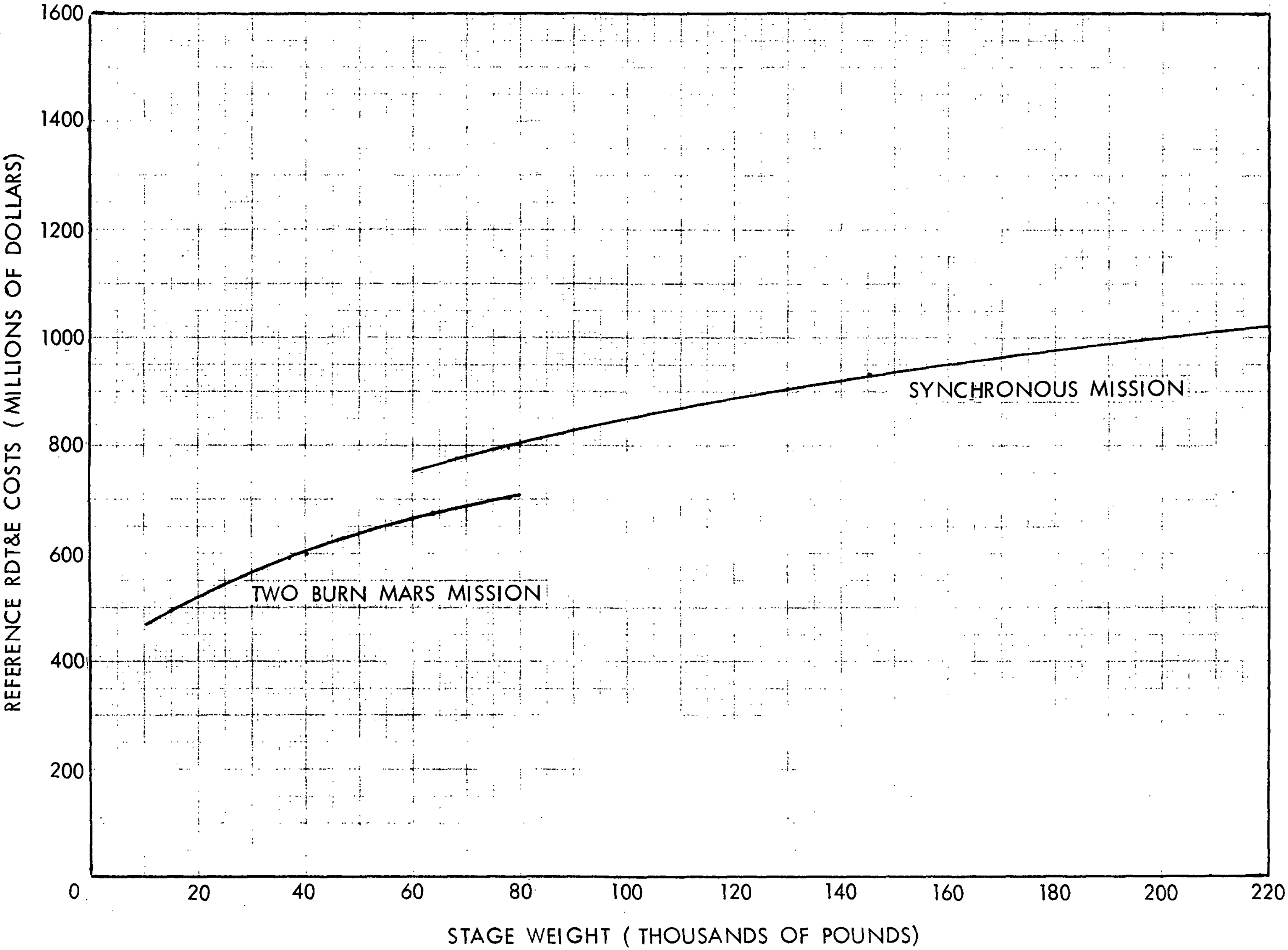

Figure 2-50. Reference RDT&E Costs

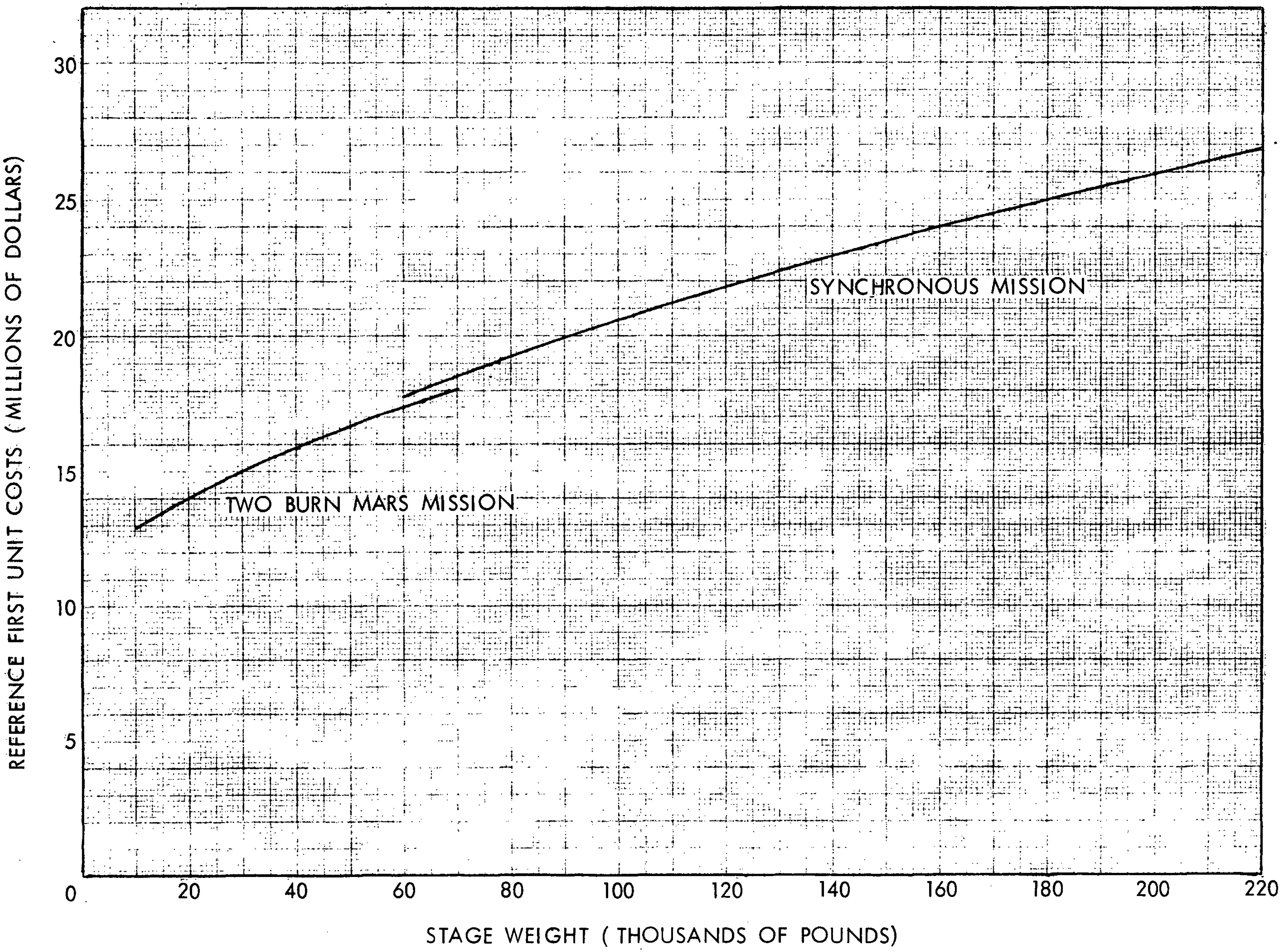

Figure 2-51. Reference First Unit Cost

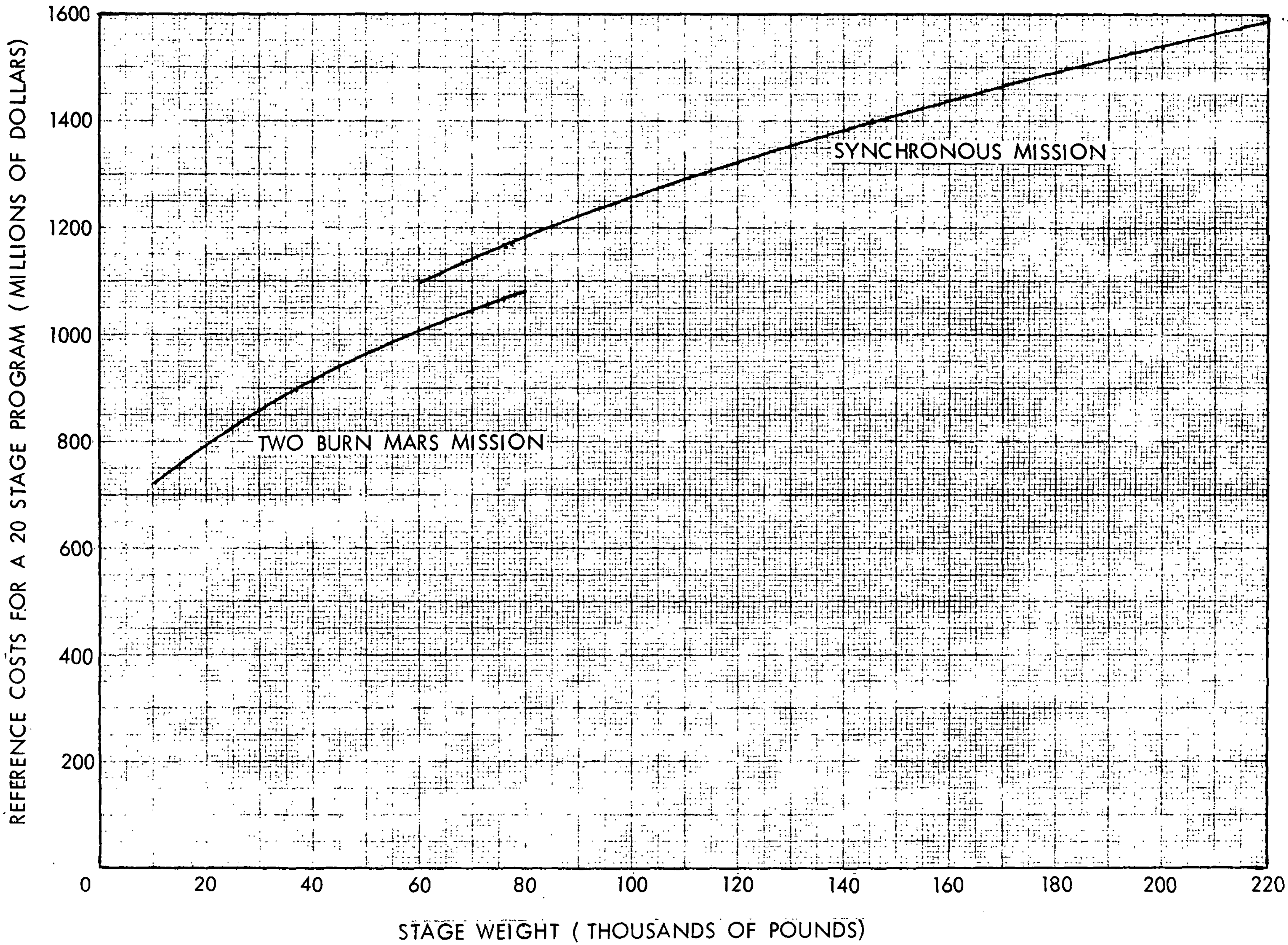

Figure 2-52. Reference Costs for a 20-Stage Program

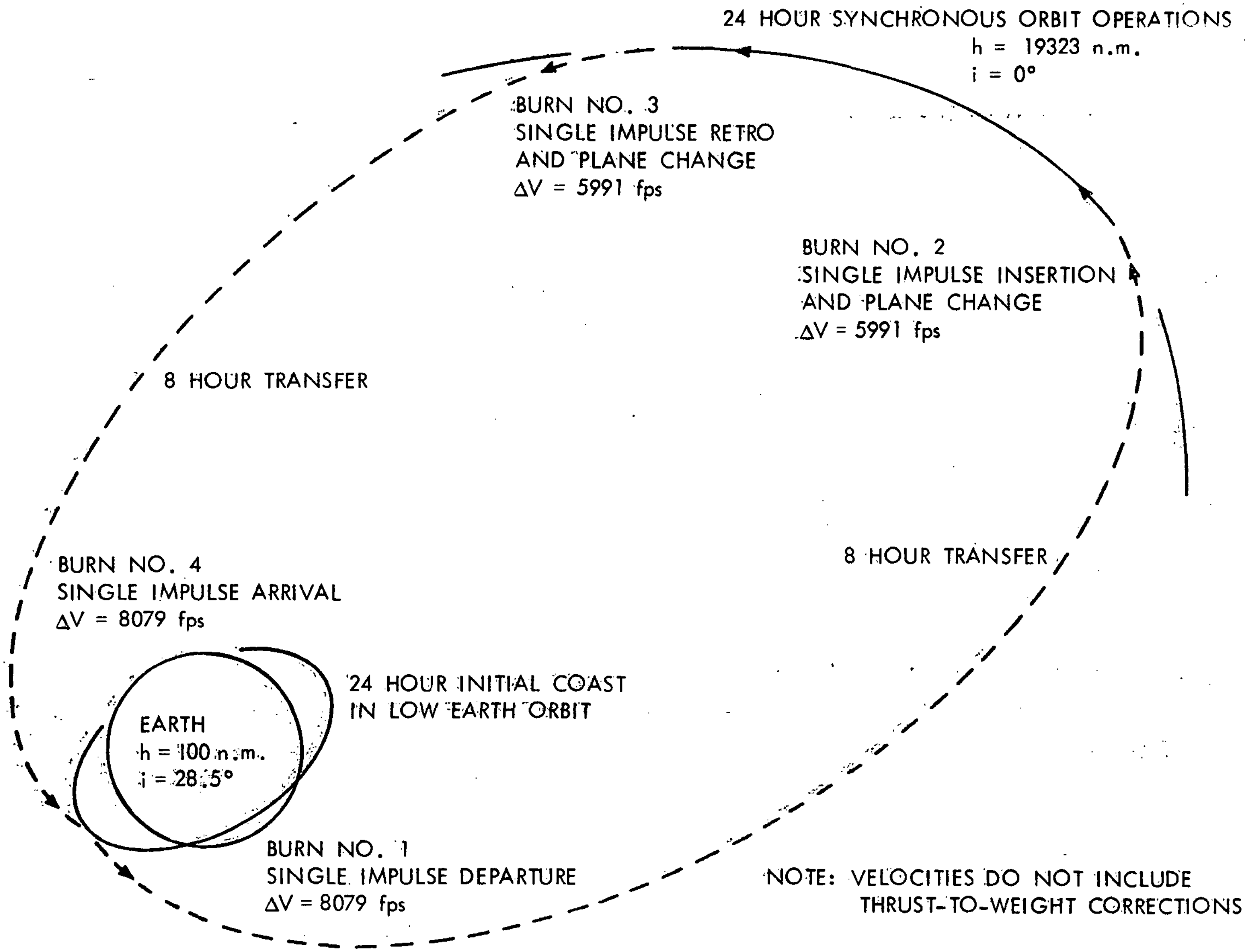

Figure 2-53. The Synchronous Mission Profile

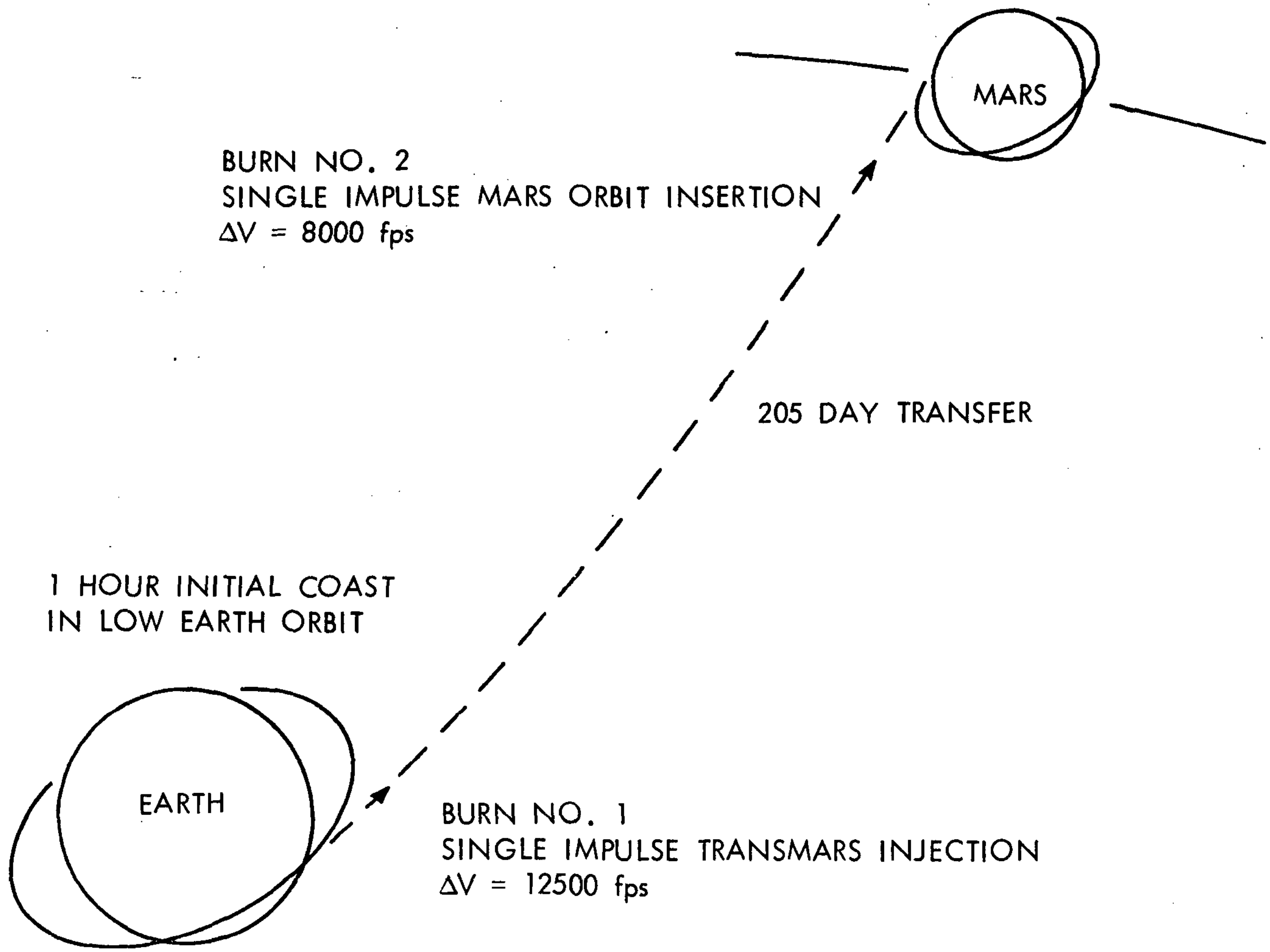

Figure 2-54. The Dual-Burn Mars Mission Profile

2.3.3 The Effect of Ultra-High Chamber-Pressure Increases

Cost variations due to increases in chamber pressure were determined for each of the following types of engine performance:

1) Theoretical specific impulse,

2) 97 Percent of theoretical specific impulse,

3) Closed-engine cycle-delivered specific impulse, and

4) Open-engine cycle-delivered specific impulse.

The results of the analyses for each of the above are discussed in the subsequent paragraphs. A comparison for the closed and open engine cycles is made in subsection 2.3.4.

2.3.3.1 Variations for Theoretical Specific Impulses

The effect of ultra-high increases in chamber pressure was analyzed by determining the variation in stage size and cost which result from the corresponding changes in theoretical specific impulse. The influence of ultra-high chamber pressure was determined for stages which had the capability of carrying payloads of up to 15,000 pounds on a round-trip synchronous mission, and 10,000 pounds on a two-burn Mars mission. The weight of stages corresponding to various payloads is shown in figure 2-55, for both the synchronous and Mars missions. These weights are presented for a chamber pressure of 2500 psi.

Figures 2-56 through 2-58 depict the savings in RDT&E, TFU and program costs, which might be realized through increased chamber pressure on a stage(s) designed for a synchronous mission. These cost savings do not reflect the cost required to develop the technology associated with the ultra-high chamber pressures. The cost savings are presented in each figure for round-trip payloads of 1000, 7500 and 15,000 pounds, which correspond to stages weights of 67-68,000, 127-130,000 and 196-200,000 pounds, respectively. Similar data, but for the two-burn Mars mission, are shown in figure 2-59 through 2-61. The stage weights which correspond to the 1000-, 5000- and 10,000-pound payloads are 14-15,000, 36-37,000 and 62-64,000 pounds, respectively.

2.3.3.2 Variations for 97 Percent Theoretical Specific Impulse

An analysis, similar to the one based on theoretical specific impulses, was conducted for the specific impulses equal to 97 percent of the theoretical value. Figure 2-62 depicts the range of stage sizes and corresponding payloads investigated during this analysis. These weights are for a stage with an engine having a chamber pressure of 2500 psi.

The cost savings in RDT&E, TFU and a 20-stage program, which might be obtained for a synchronous mission stage through increased chamber pressure, are presented in figures 2-63, 2-64, and 2-65, respectively. The cost saving presented in these figures does not reflect the cost associated with the technology required to increase the chamber pressure. The cost savings are presented in each figure for stages capable of carrying round-trip

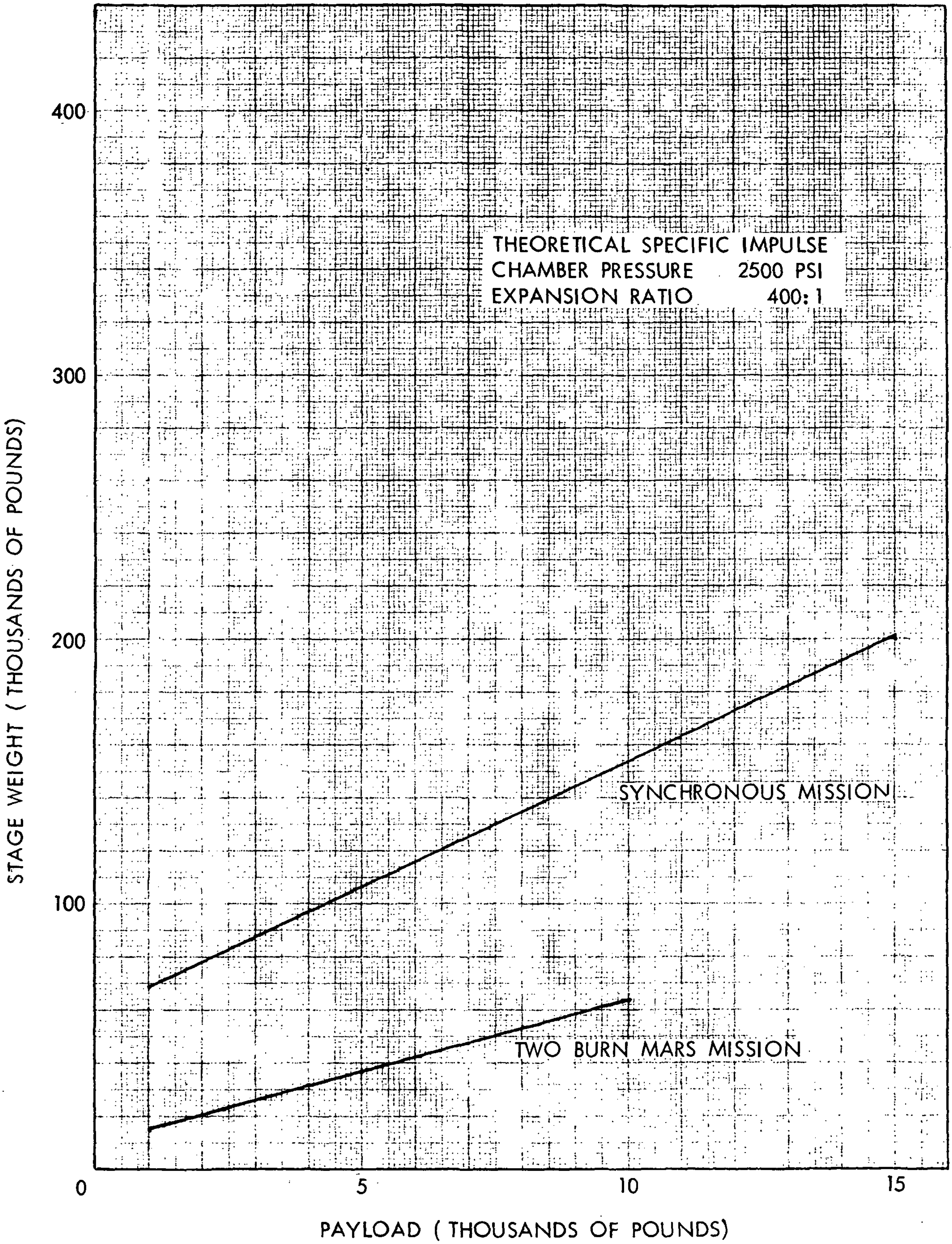

Figure 2-55. Stage Weights (Theoretical Specific Impulse)

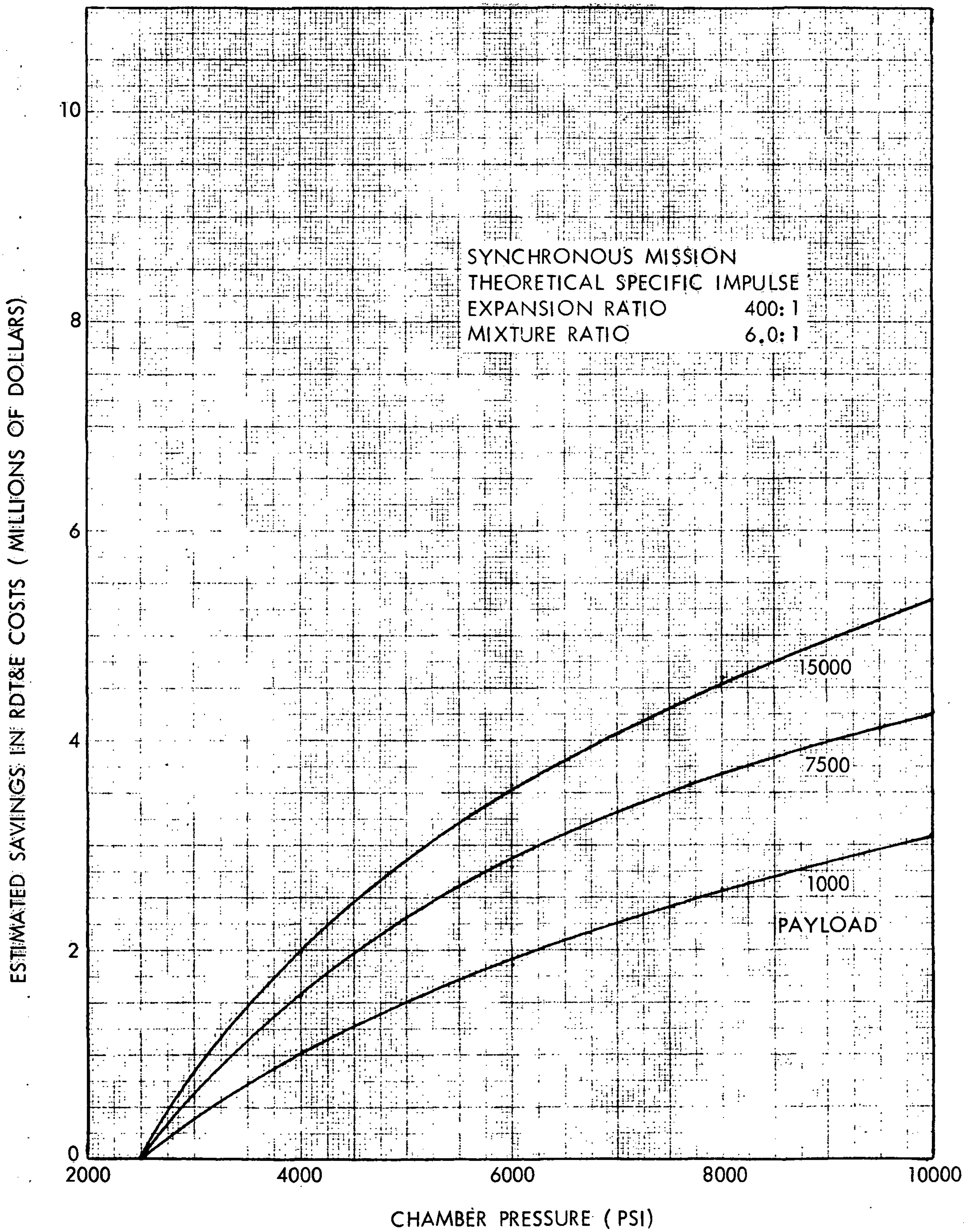

Figure 2-56. RDT&E Savings for a Synchronous Mission Stage (Theoretical Specific Impulse)

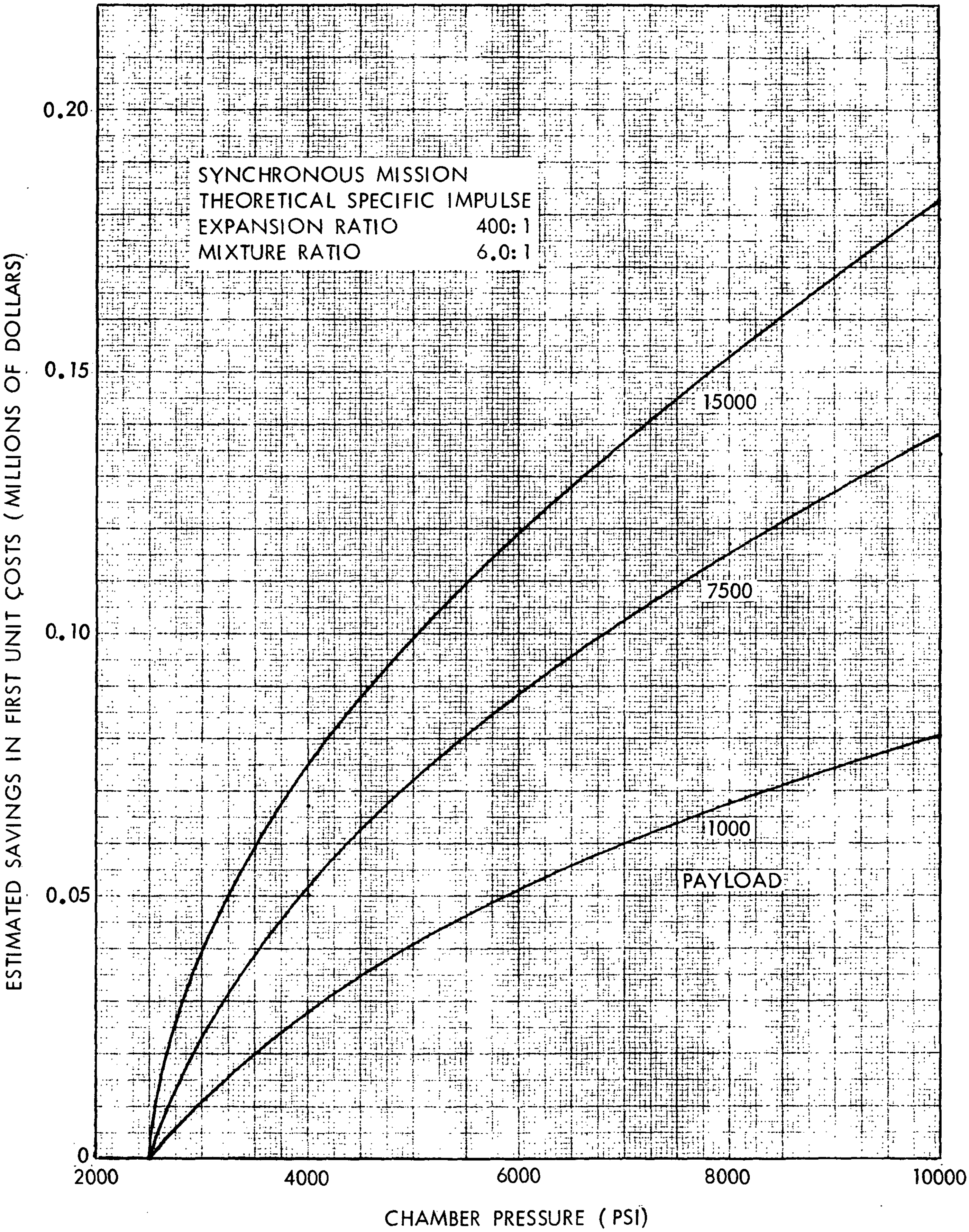

Figure 2-57. TFU Savings for a Synchronous Mission Stage (Theoretical Specific Impulse)

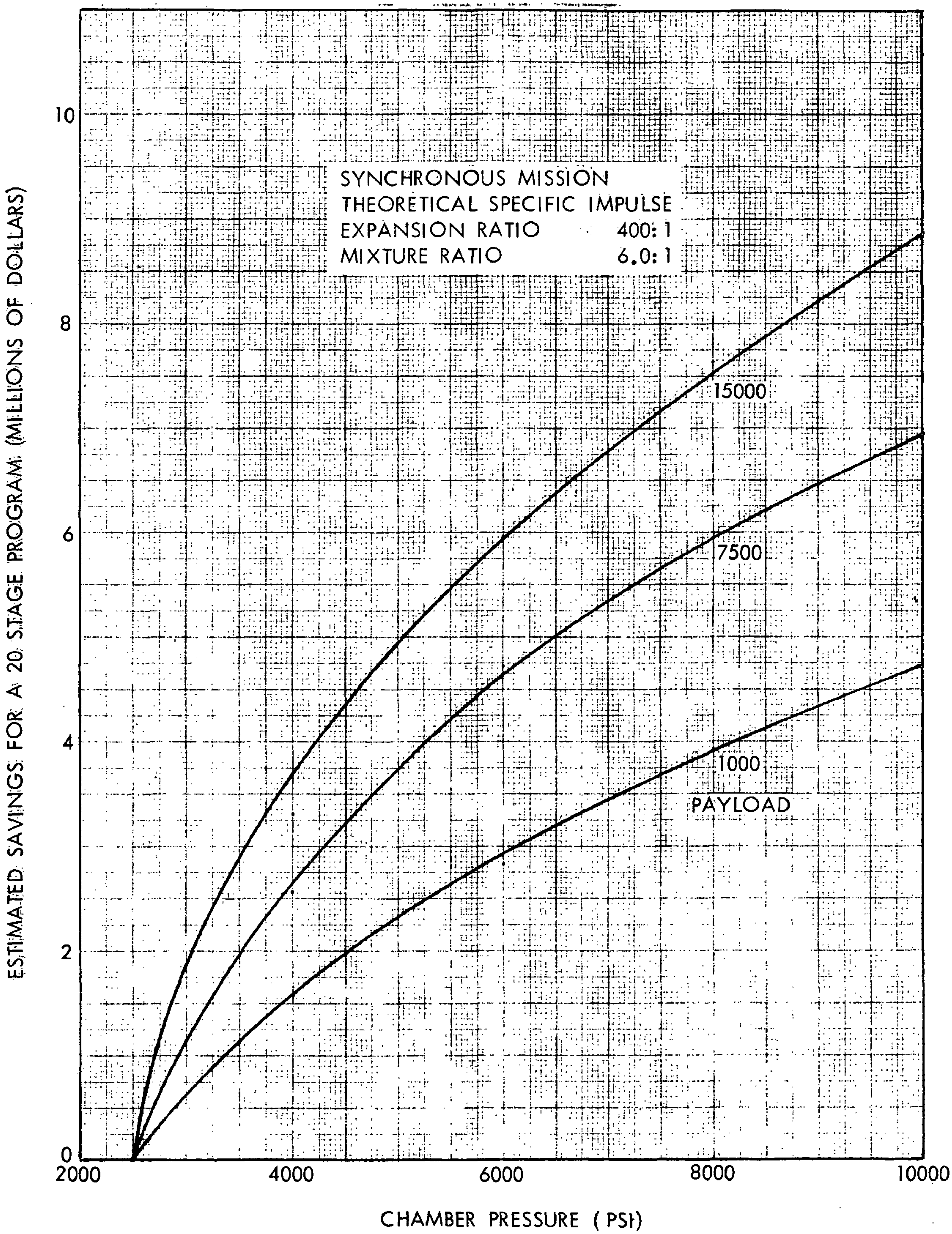

Figure 2-58. Program Saving for a Synchronous Mission Stage (Theoretical Specific Impulse)

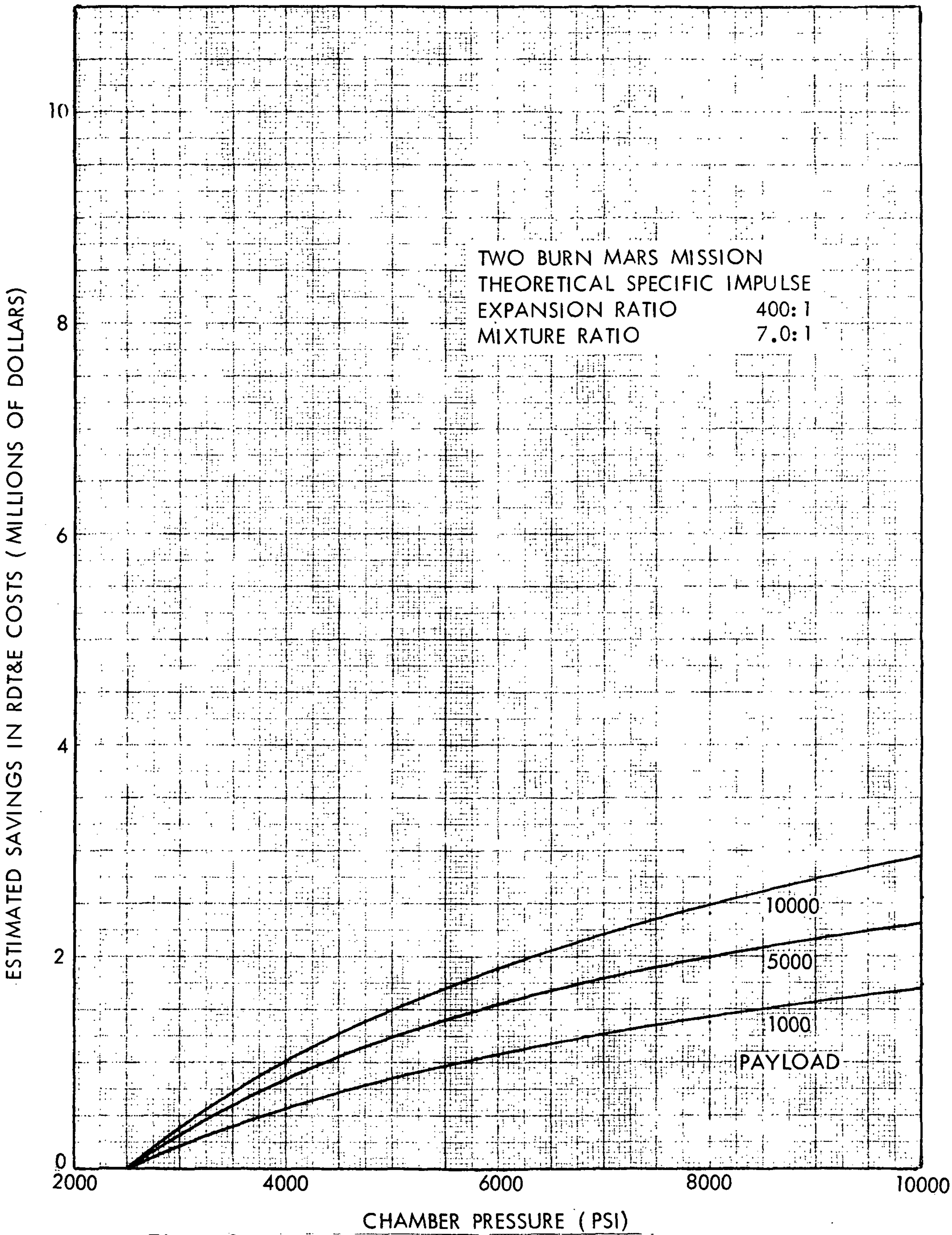

Figure 2-59. RDT&E Savings for a Mars Mission Stage (Theoretical Specific Impulse)

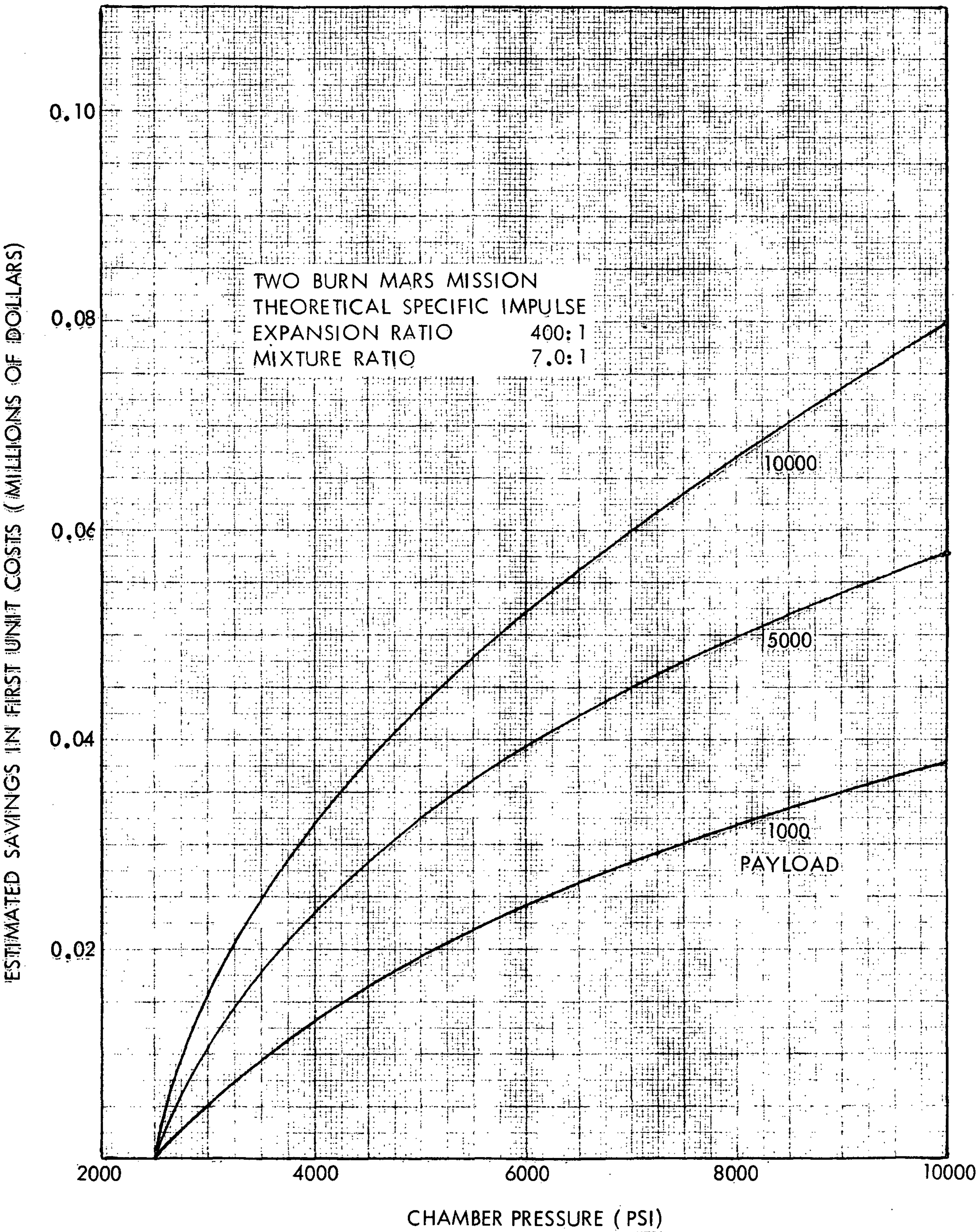

Figure 2-60. TFU Savings for a Mars Mission Stage (Theoretical Specific Impulse)

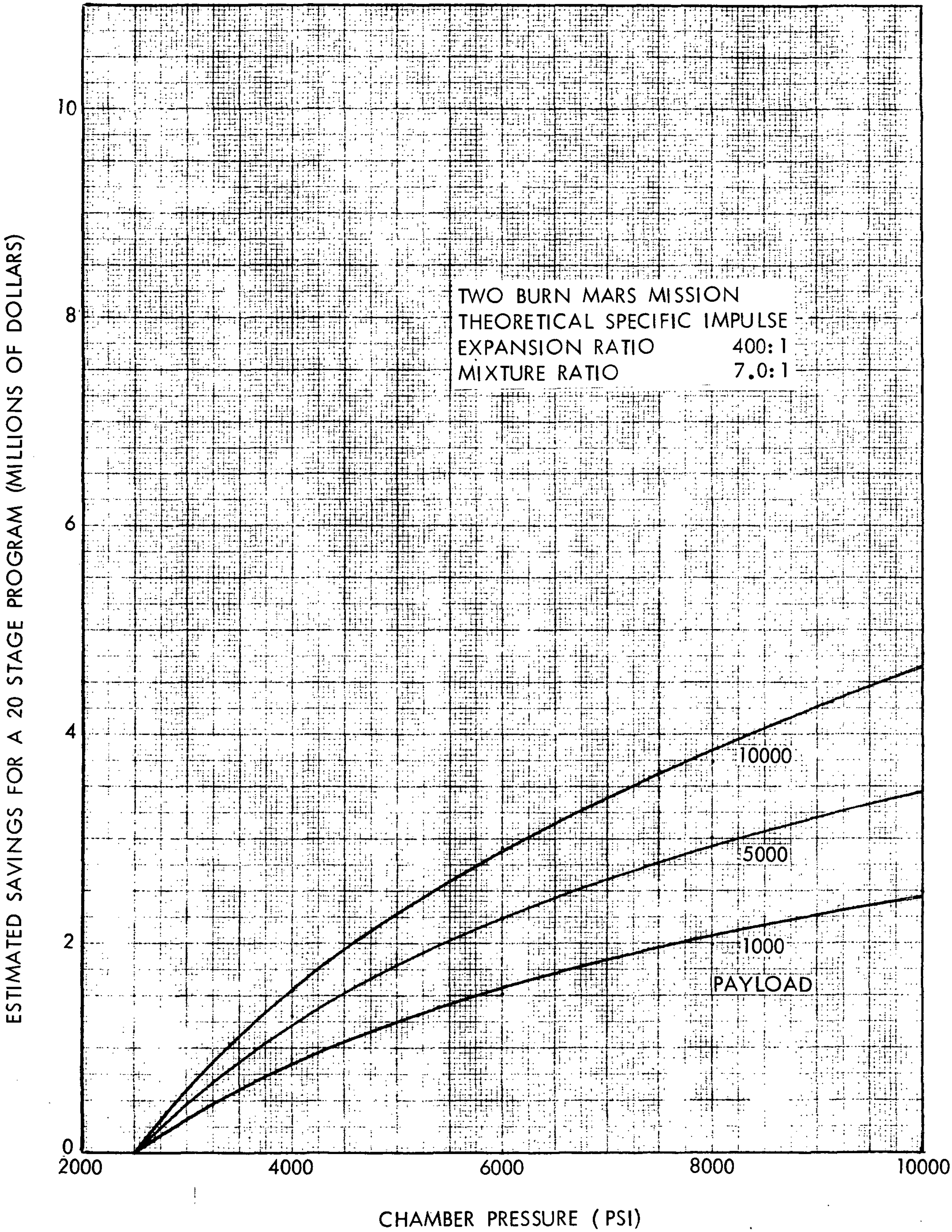

Figure 2-61. Program Savings for a Mars Mission Stage
(Theoretical Specific Impulse)

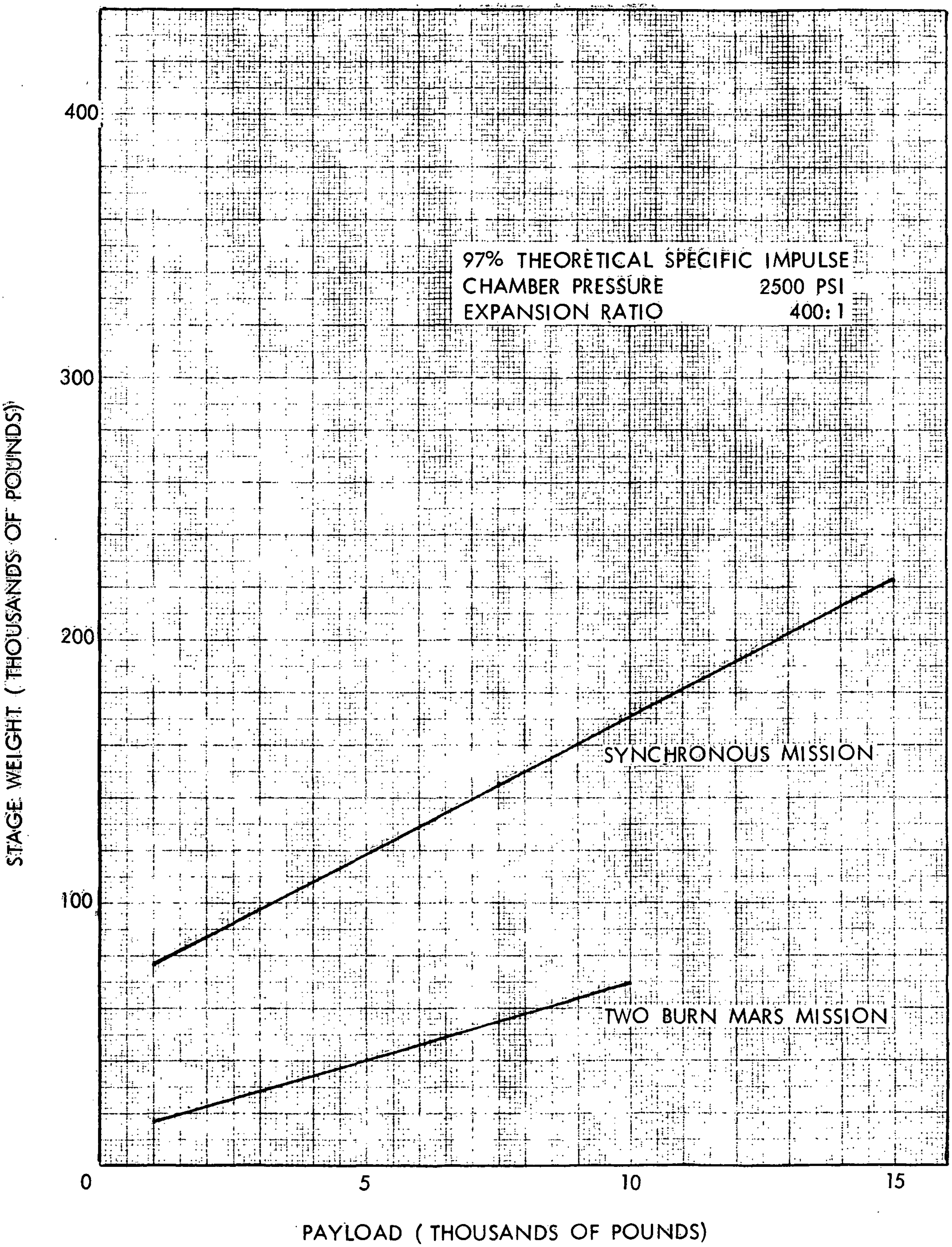

Figure 2-62. Stage Weights (97 Percent Theoretical Specific Impulse)

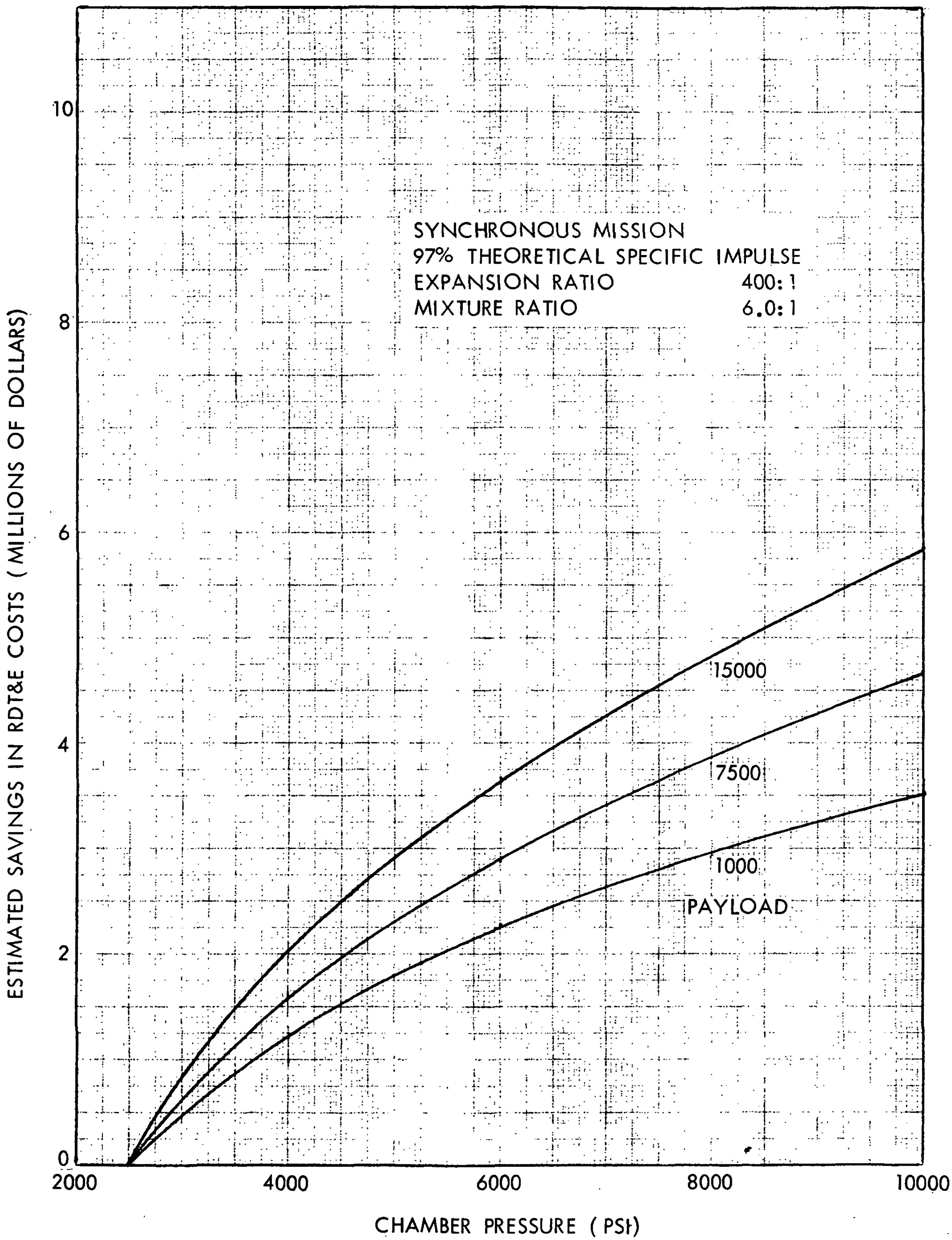

Figure 2-63. RDT&E Savings for a Synchronous Mission Stage (97 Percent Theoretical Specific Impulse)

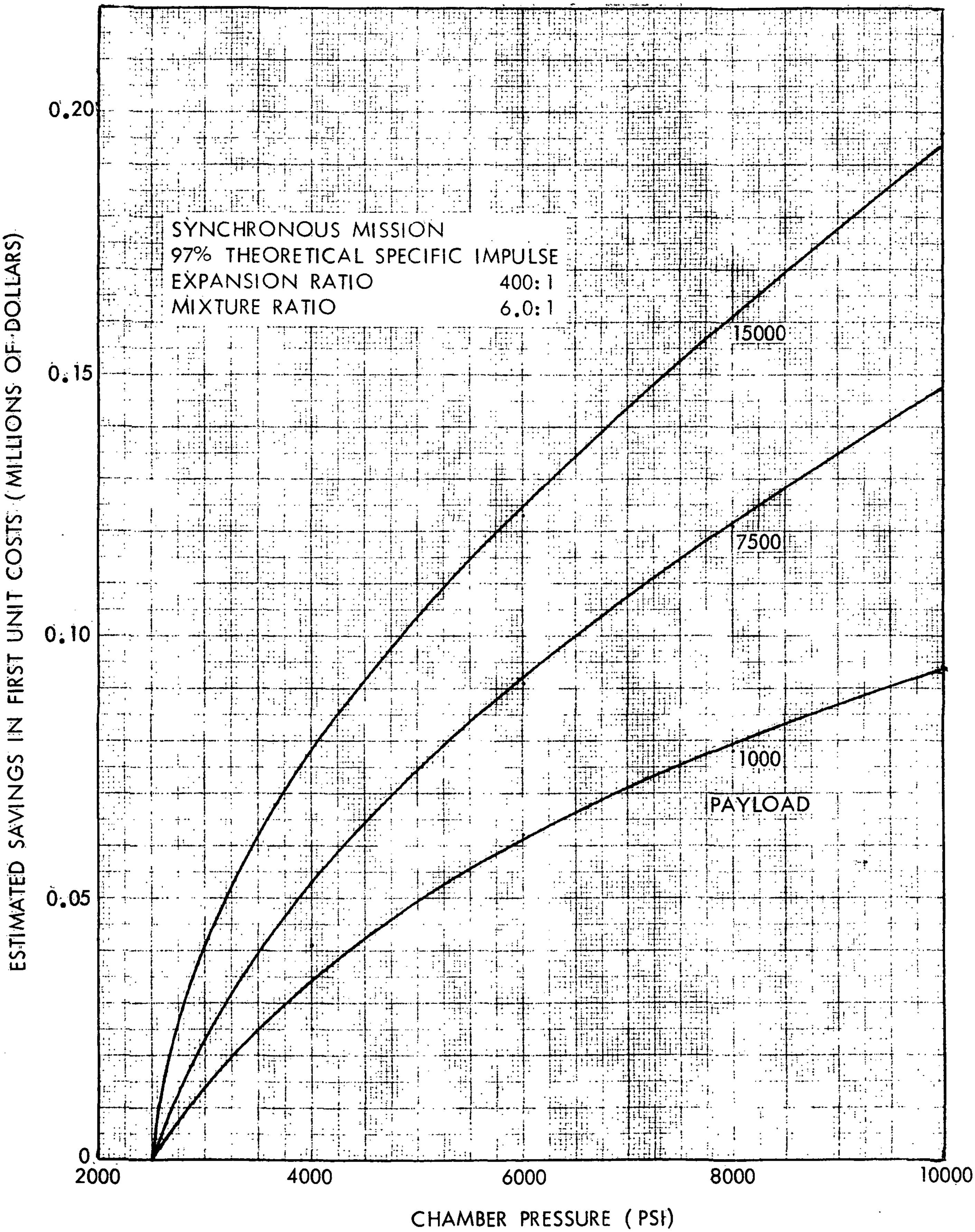

Figure 2-64. TFU Savings for a Synchronous Mission Stage (97 Percent Theoretical Specific Impulse)

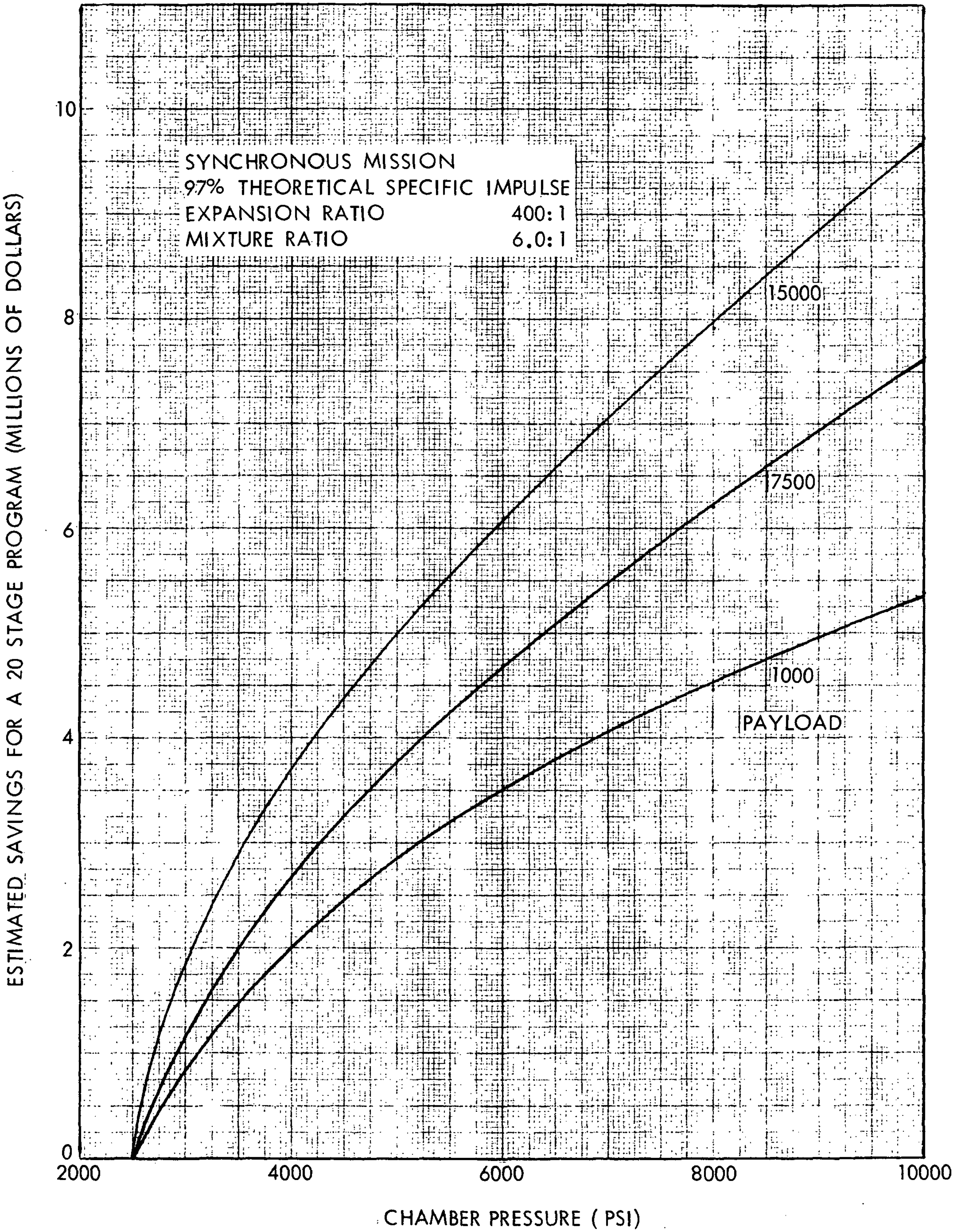

Figure 2-65. Program Savings for a Synchronous Mission Stage
(97 Percent Theoretical Specific Impulse)

payloads of 1000, 7500 and 15,000 pounds. The corresponding stage weights are 74-76,000, 142-146,000 and 219-224,000 pounds.

A similar set of data for the two-burn Mars mission is shown in figures 2-66 through 2-68. The stage weights which correspond to the 1000-, 5000- and 10,000-pound payloads are 16-17,000, 39-40,000 and 68-70,000 pounds, respectively.

2.3.3.3 Variations for a Closed-Engine Cycle

In addition to the effects of theoretical specific impulses, analyses were conducted to determine the effect that ultra-high increases in chamber pressure would have on the delivered specific impulse of the stages, and hence the stage's weight and cost. Two different engine cycles were investigated. The first was the pump-fed closed-engine cycle, such as a topping or expansion cycle. Although the pump-fed closed-cycle engines are limited, by a turbo-machinery efficiency of approximately 80 percent, to a maximum chamber pressure of 4000-5000 psi, chamber pressures up to 10,000 psi were studied (see appendix B). The results of this analysis for the synchronous (0-15,000-lb payloads) and Mars (0-10,000-lb payloads) missions are depicted in figures 2-69 through 2-75. The stage weights corresponding to the various payloads are presented for both the synchronous and Mars missions in figure 2-69. These stages were designed with a closed-engine cycle having a chamber pressure of 2500 psi.

The cost savings for RDT&E, TFU and program, which might be realized through increased chamber pressure on a stage(s) designed for a synchronous mission, are presented in figures 2-70, 2-71 and 2-72, respectively. These cost savings do not reflect the cost required to develop the technology required to obtain these ultra-high chamber pressures. The cost savings are illustrated in each figure for round-trip payloads of 1000, 7500 and 15,000 pounds, which correspond to stage weights of 75-78,000, 144-148,000 and 221-228,000 pounds, respectively.

Similar data, but for the two-burn Mars missions, are shown in figures 2-73 through 2-75. The stage weights which correspond to the 1000-, 5000- and 10,000-pound payloads are 16-17,000, 39-41,000 and 68-70,000 pounds, respectively.

2.3.3.4 Variations for an Open-Engine Cycle.

The second type of engine cycle studied was the pump-fed open-engine cycle. Although this cycle is capable of operating at chamber pressures higher than the previously discussed closed cycle (see section 2.3.3.3), the specific impulse efficiency of the open-cycle engine begins to decay rapidly with increased chamber pressure. Hence, for stages using engines with open cycles, the stage weight and costs increase with increasing chamber pressures. The results of the open-engine cycle investigation are presented in figures 2-76 through 2-82. Figure 2-76 depicts the weights of stages using an open-cycle engine with a 2500 psi chamber pressure, as a function of payload.

Presented in figures 2-77, 2-78 and 2-79 are the RDT&E, TFU and program negative cost savings, respectively, which would be incurred for a synchronous mission stage through increased chamber pressure. These indicated increased costs associated with the open-cycle engine do not include the

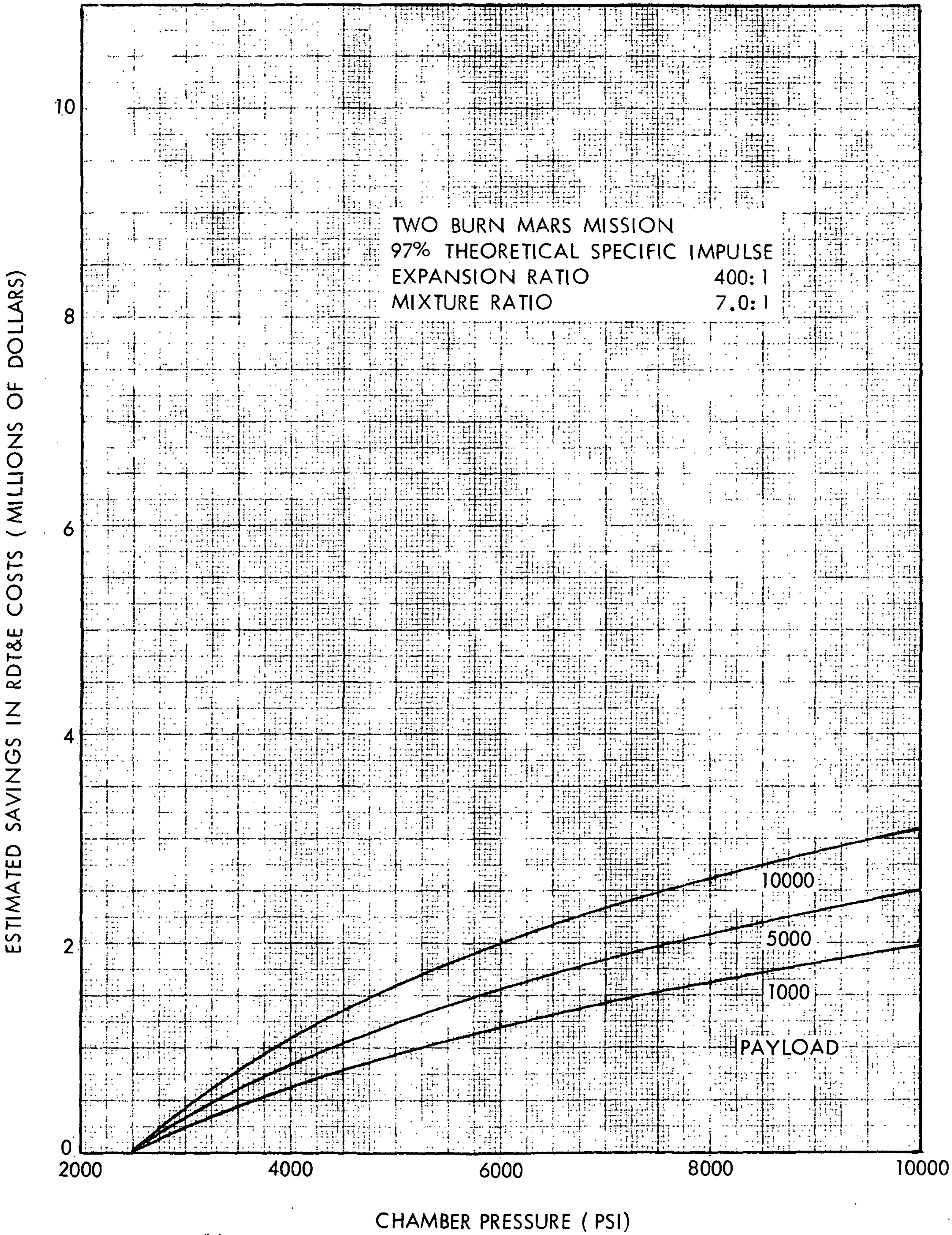

Figure 2-66. RDT&E Savings for a Mars Mission Stage (97 Percent Theoretical Specific Impulse)

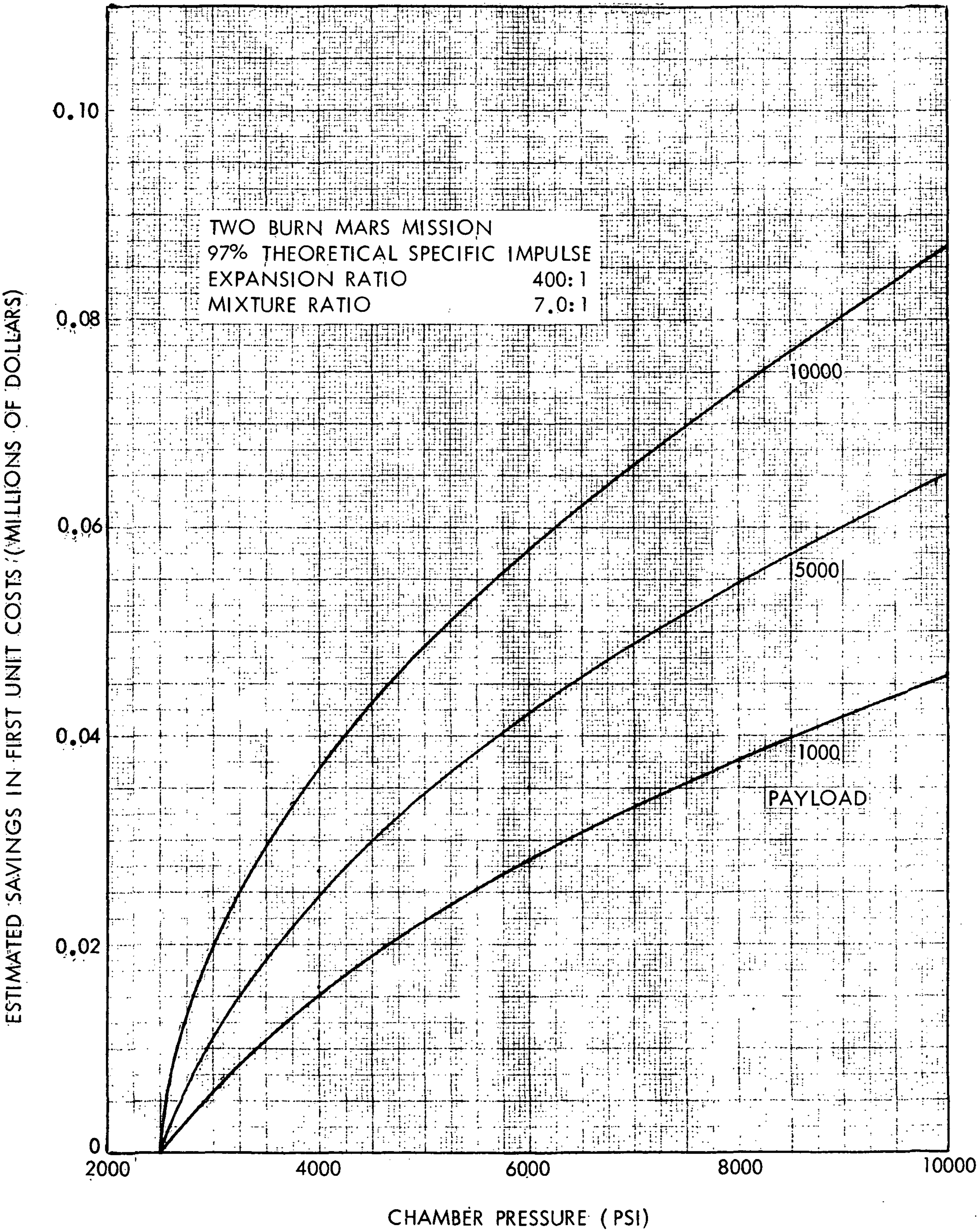

Figure 2-67. TFU Savings for a Mars Mission Stage (97 Percent Theoretical Specific Impulse)

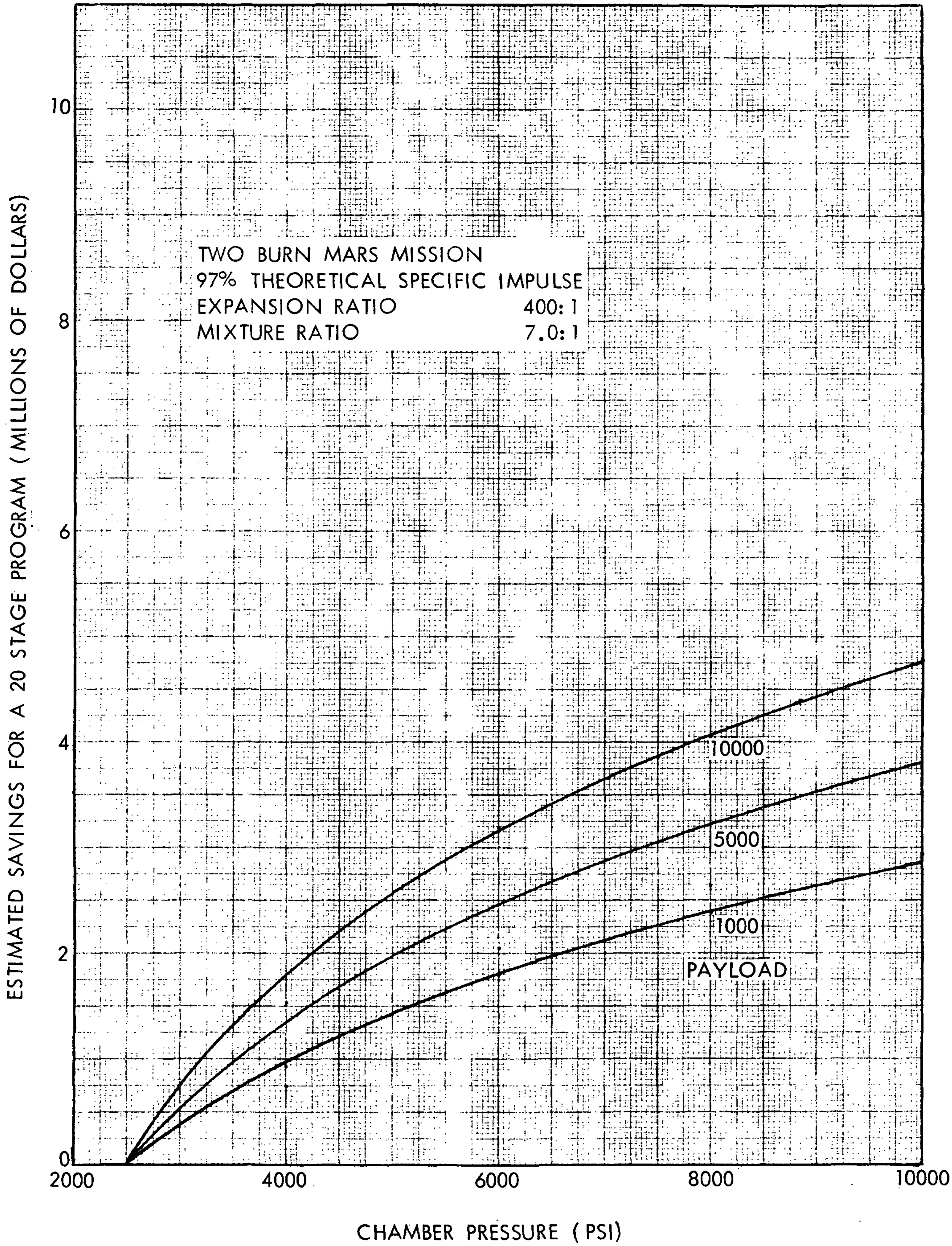

Figure 2-68. Program Savings for a Mars Mission Stage
(97 Percent Theoretical Specific Impulse)

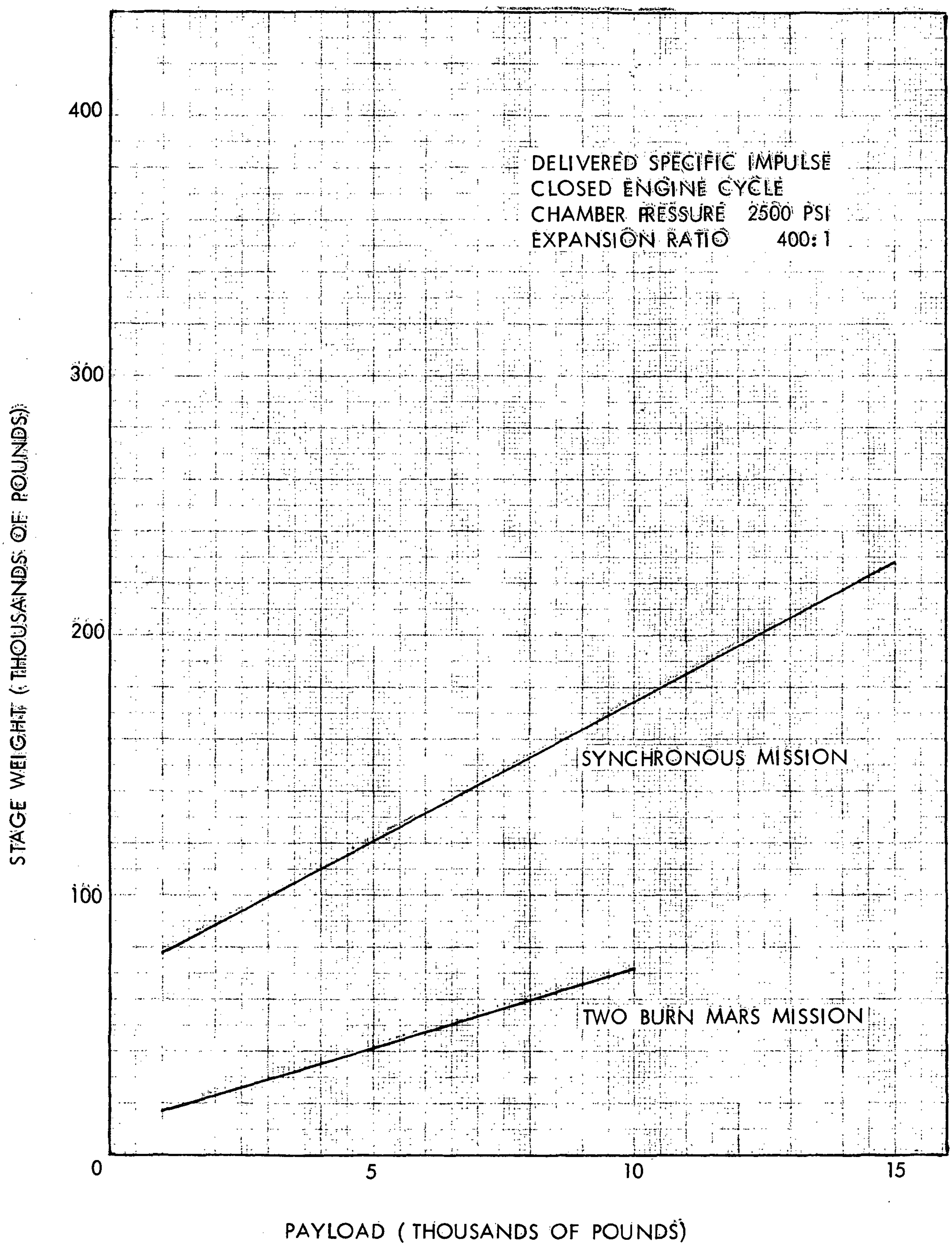

Figure 2-69. Stage Weight (Delivered Closed Cycle Specific Impulse)

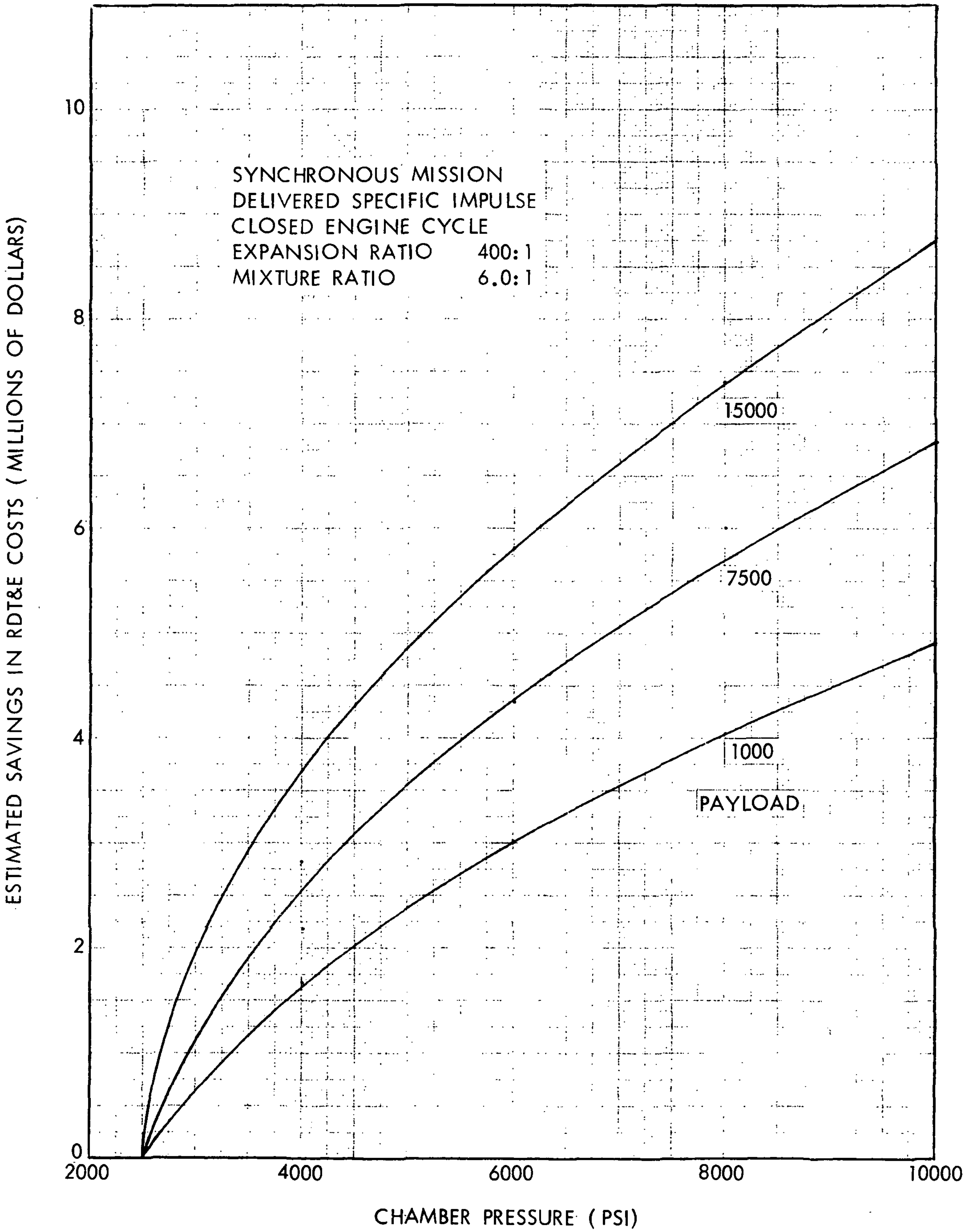

Figure 2-70. RDT&E Savings for a Synchronous Mission Stage
(Delivered Closed Cycle Specific Impulse)

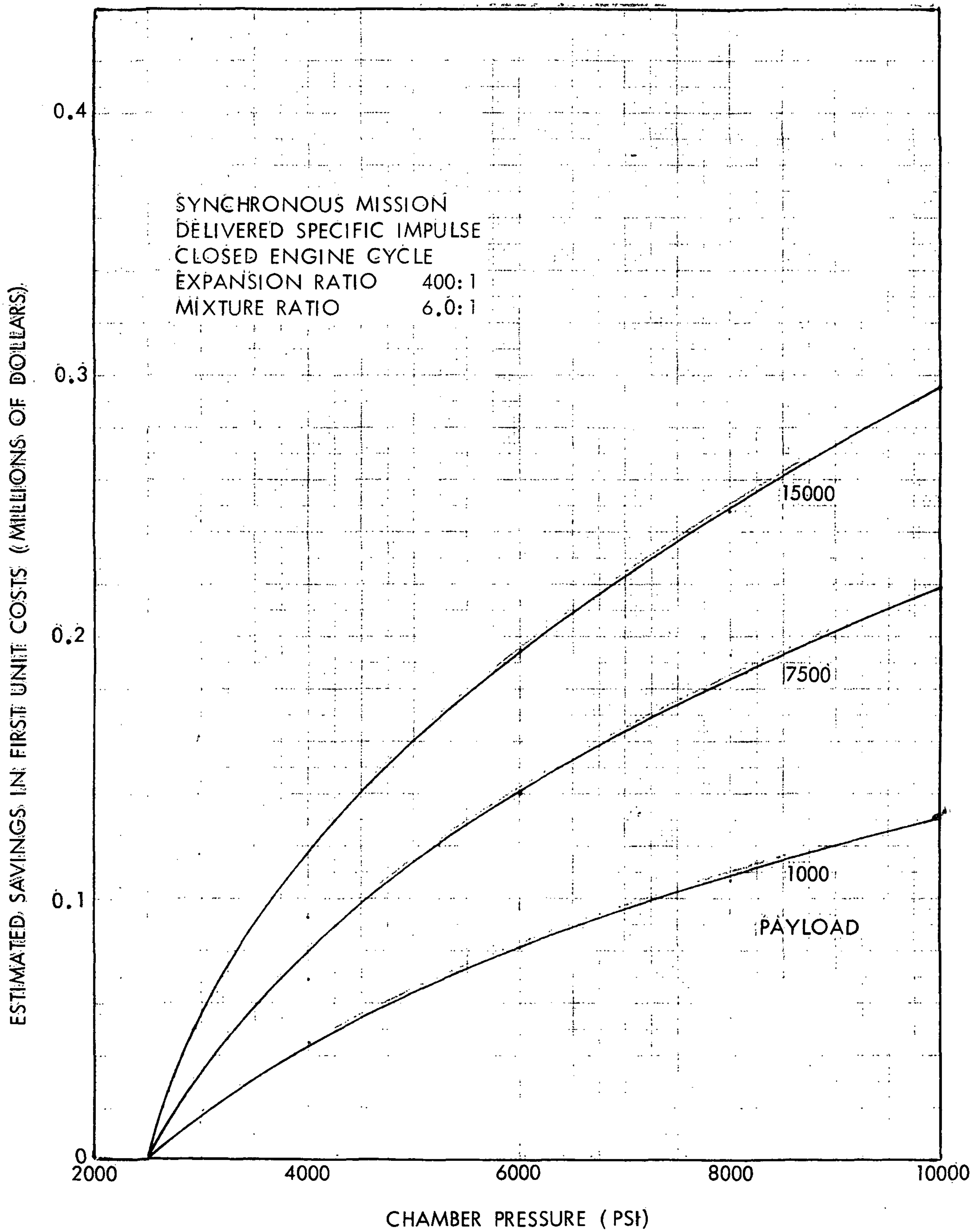

Figure 2-71. TFU Savings for a Synchronous Mission Stage (Delivered Closed Cycle Specific Impulse)

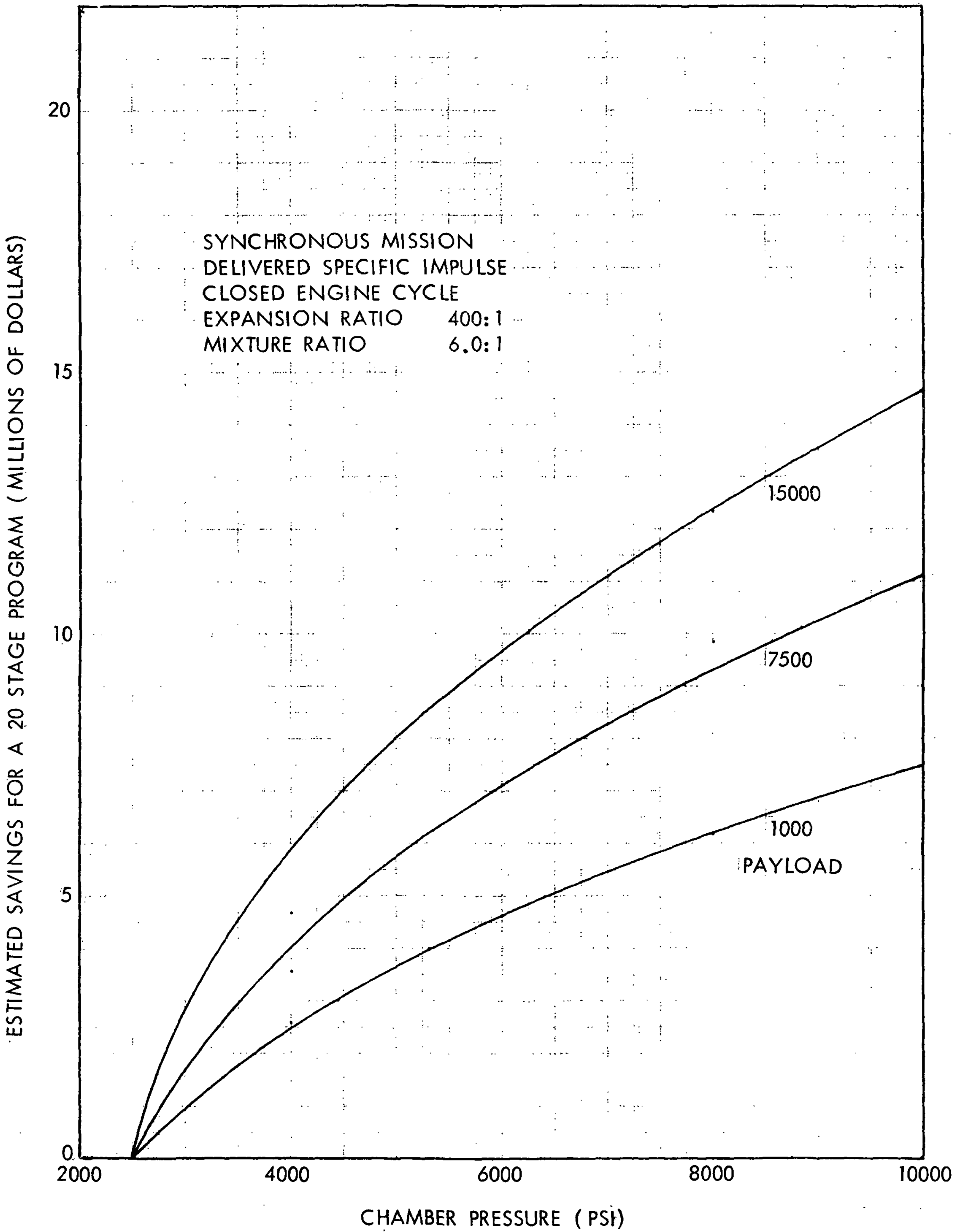

Figure 2-72. Program Savings for a Synchronous Mission Stage (Delivered Closed Cycle Specific Impulse)

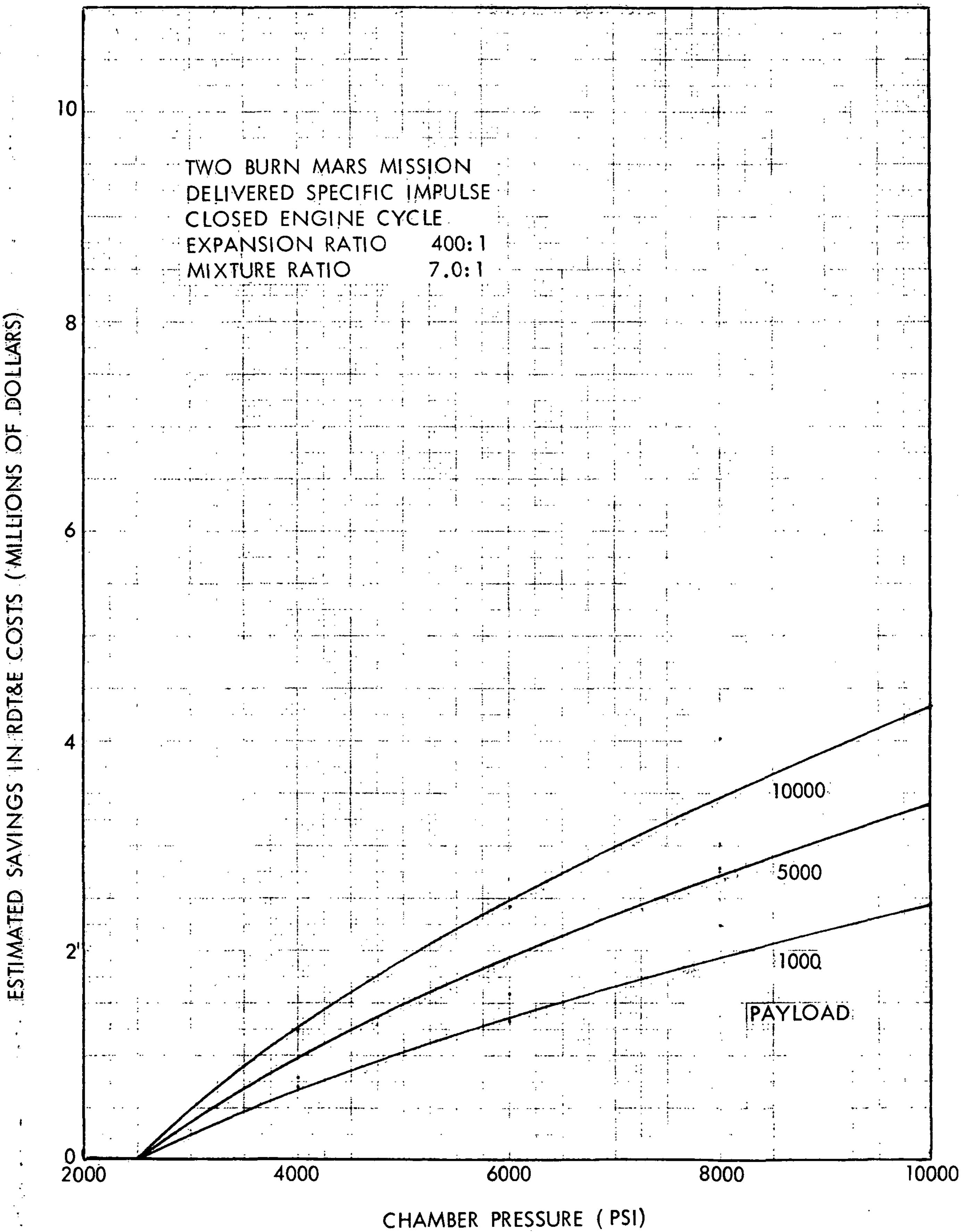

Figure 2-73. RDT&E Savings for a Mars Mission Stage (Delivered Closed Cycle Specific Impulse)

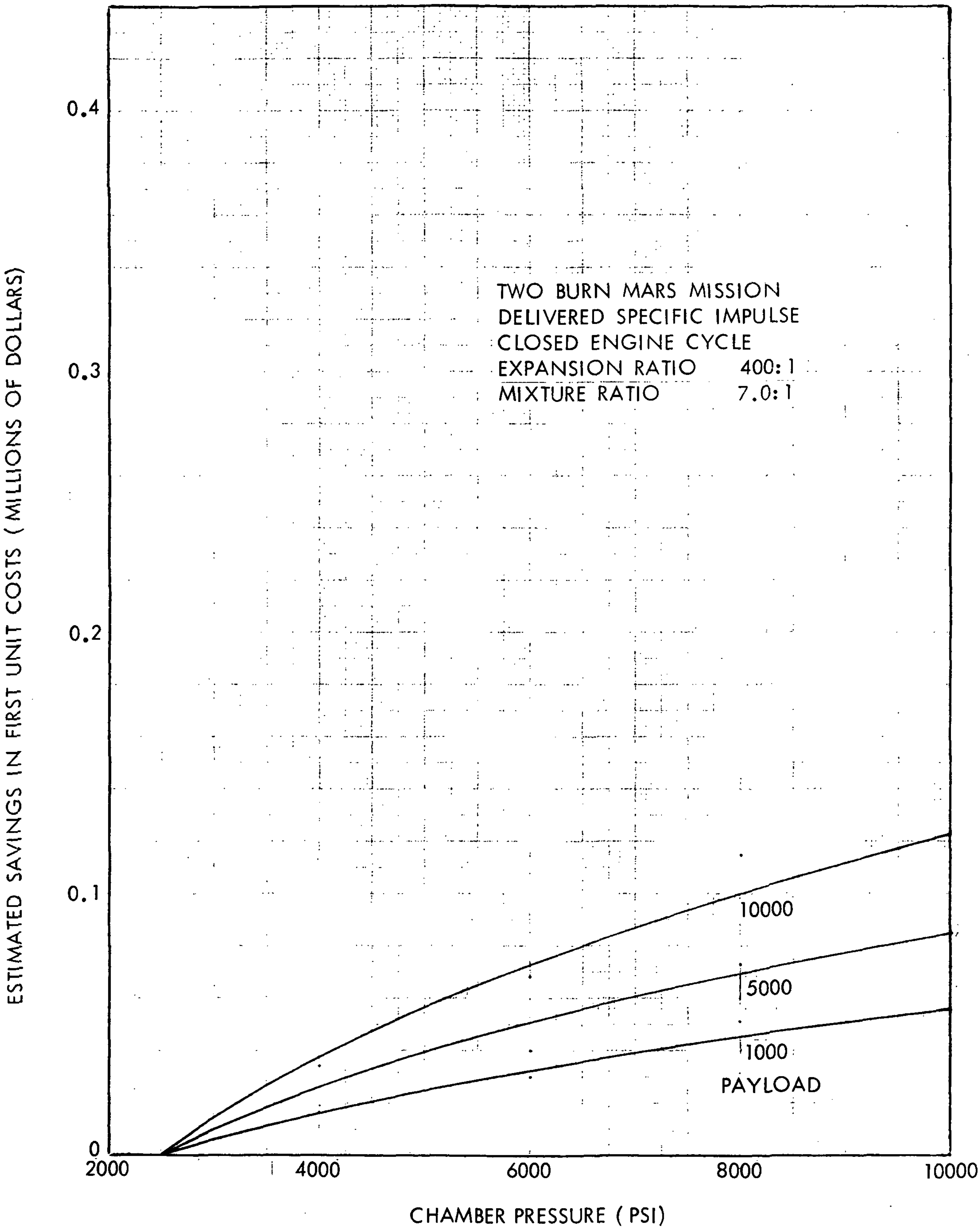

Figure 2-74. TFU Savings for a Mars Mission Stage (Delivered Closed Cycle Specific Impulse)

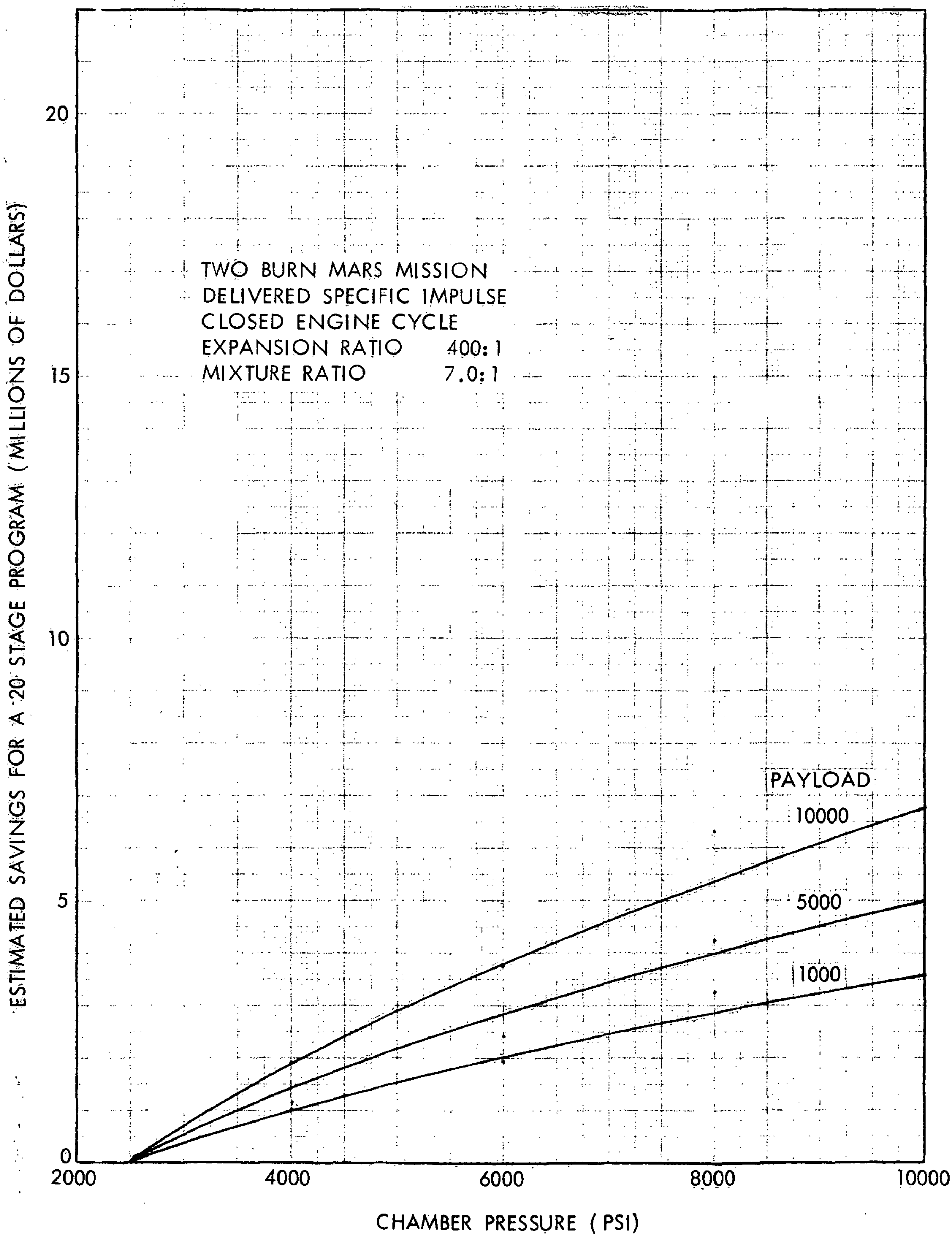

Figure 2-75. Program Saving for a Mars Mission Stage (Delivered Closed Cycle Specific Impulse)

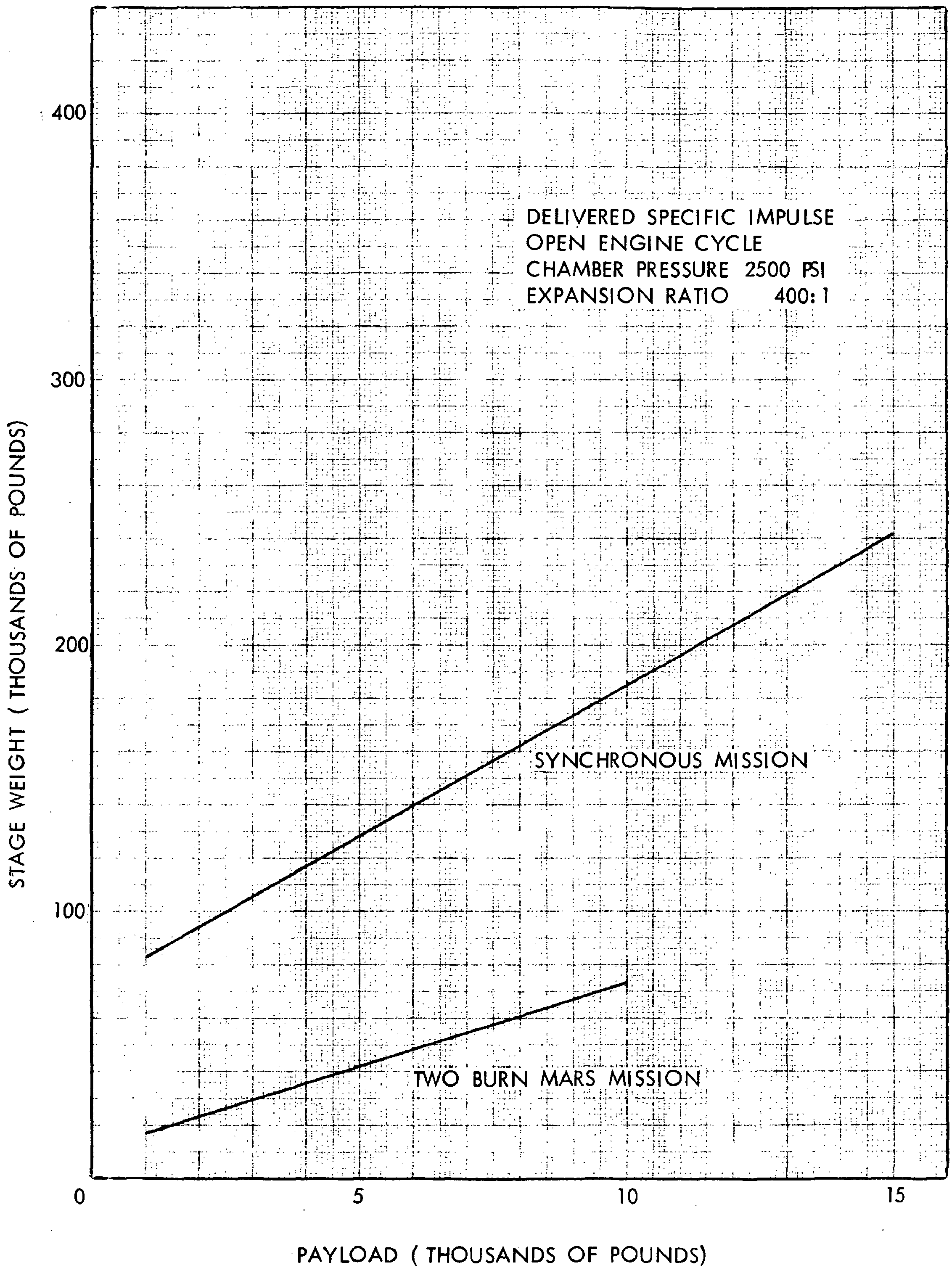

Figure 2-76. Stage Weights (Delivered Open-Cycle Specific Impulse)

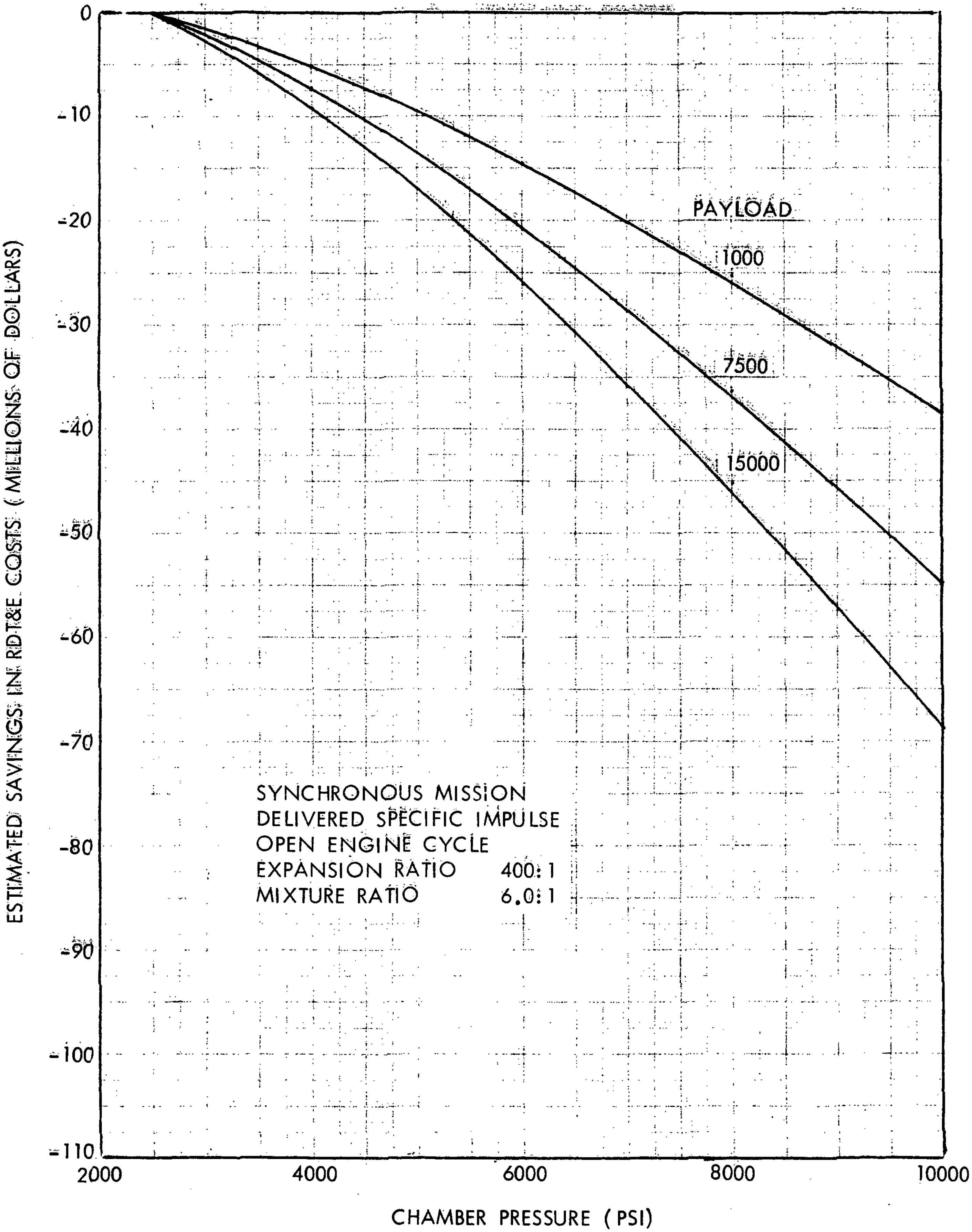

Figure 2-77. RDT&E Savings for a Synchronous Mission Stage (Delivered Open-Cycle Specific Impulse)

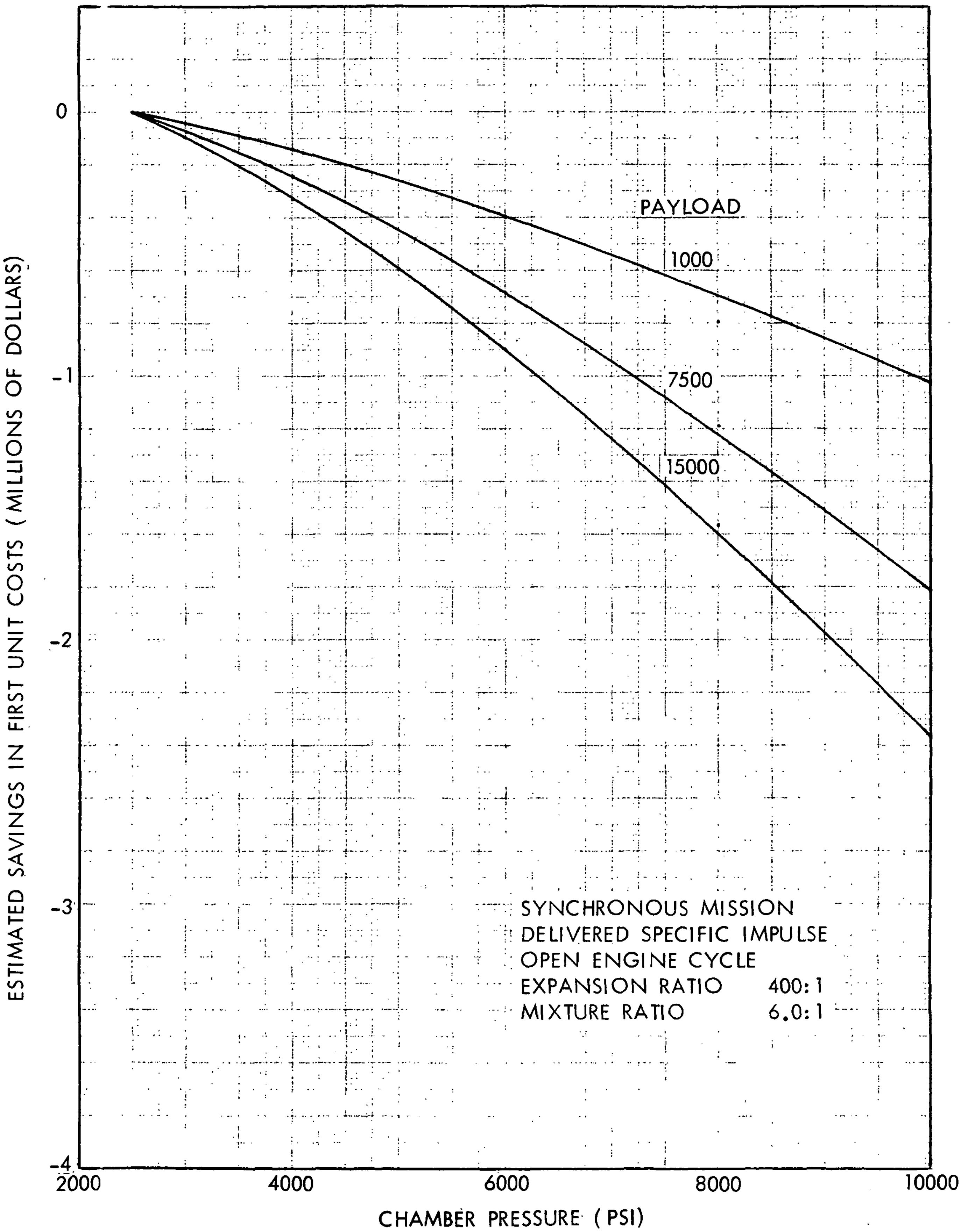

Figure 2-78. TFU Savings for a Synchronous Mission Stage (Delivered Open-Cycle Specific Impulse)

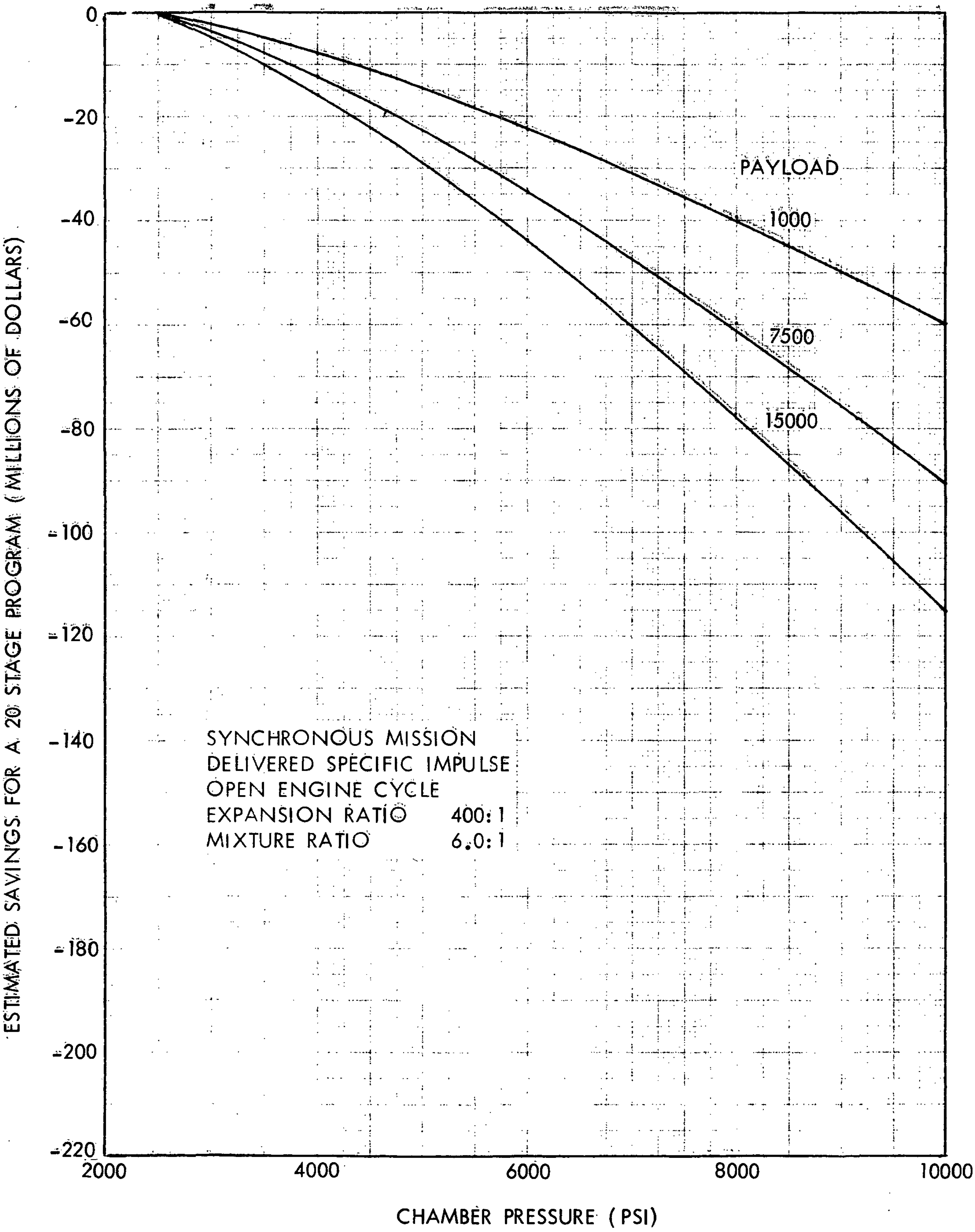

Figure 2-79. Program Savings for a Synchronous Mission Stage (Delivered Open-Cycle Specific Impulse)

technology costs associated with obtaining the ultra-high chamber pressures. The delta costs are presented in each figure for stages capable of carrying round-trip payloads of 1000, 7500 and 15,000 pounds. The stage weights which correspond to these payloads are in the order of 82-102,000, 157-195,000 and 197-242,000 pounds, respectively.

For the Mars mission, a similar set of data is presented in figures 2-80 through 2-82. The variations in cost are presented for stages designed to carry 1000-, 5000- and 10,000-pound payloads, which correspond to stage weights of 17-20,000, 42-48,000 and 73-83,000 pounds.

2.3.4 Comparison of the Closed- and Open-Engine Cycles

The most probably effect that ultra-high increases in chamber pressure would have on stage size and cost is best illustrated by comparing the results of the analyses of the open- and closed-engine cycles. Figures 2-83 through 2-86 show the stage weight variations and the differences in cost for stages designed to carry a 15,000-pound payload (round trip) on the synchronous mission. This size stage was one of the largest investigated during this study. Because the open-cycle engine has a lower delivered specific impulse (7 seconds) at 2500 psi than the closed-cycle engine, the stages using an open-engine cycle weigh more at the 2500-psi reference chamber pressure. This is why the stage weight curves for the two cycles do not intersect at 2500 psi in figure 2-83.

The higher stage weights also result in higher costs for the open-engine cycle stages at the reference chamber pressure. The RDT&E, TFU and program savings depicted in figures 2-84, 2-85 and 2-86, respectively, indicate a higher cost associated with the open-engine cycle. The cost associated with developing the ultra-high chamber pressure technology are not reflected in these cost comparisons. The data presented in these figures are referenced to the cost of a stage with a 2500-psi closed-cycle engine.

A similar set of data is presented for a Mars mission stage in figures 2-87 through 2-90. This stage, which was designed to carry a 1000-pound payload, was one of the smallest considered in this study.

In both instances, the large synchronous stage and the small Mars stage, the use of the closed-engine cycle yields stages which are smaller and less expensive. However, as indicated in the engine performance data (see appendix B), the closed-engine cycle is limited by turbo-machinery efficiencies to maximum chamber pressures of 4000-5000 psi. Present state-of-the-art turbo-machinery has efficiencies in the order of 80 percent, which correspond to a chamber pressure of 4200 psi. Hence, for all practical purposes, open-engine cycles must be selected for all chamber pressures in excess of 4000-5000 psi. But, because the delivered specific impulse of an open-cycle engine operating at these chamber pressures is 15 to 17 seconds less than that of the closed-cycle engine, stages using open-cycle engines will weigh and cost more. In fact, as chamber pressure is increased further, the stage using an open-cycle engine becomes even more costly because the delivered specific impulse drops rapidly with chamber pressure increase.

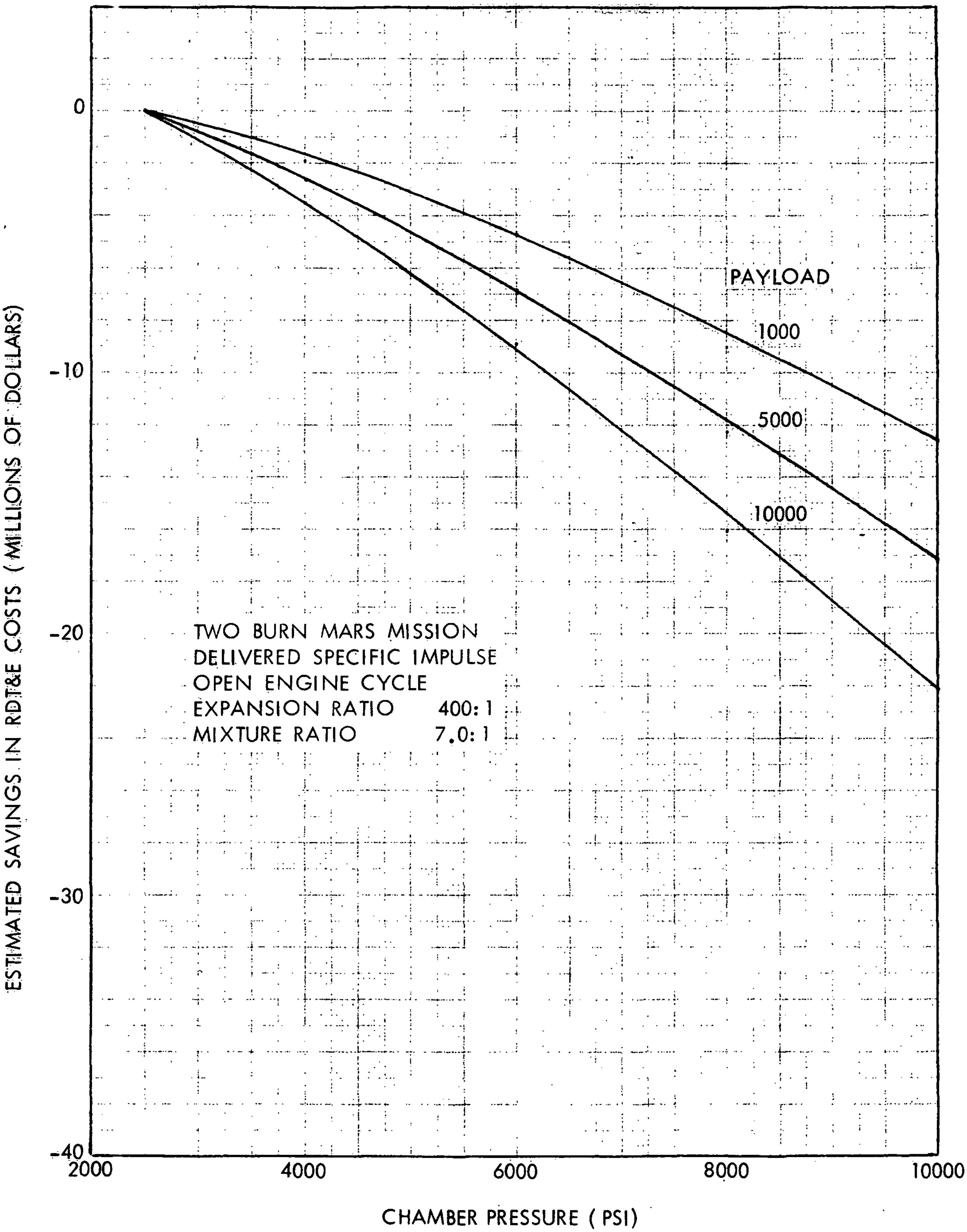

Figure 2-80. RDT&E Savings for a Mars Mission Stage (Delivered Open-Cycle Specific Impulse)

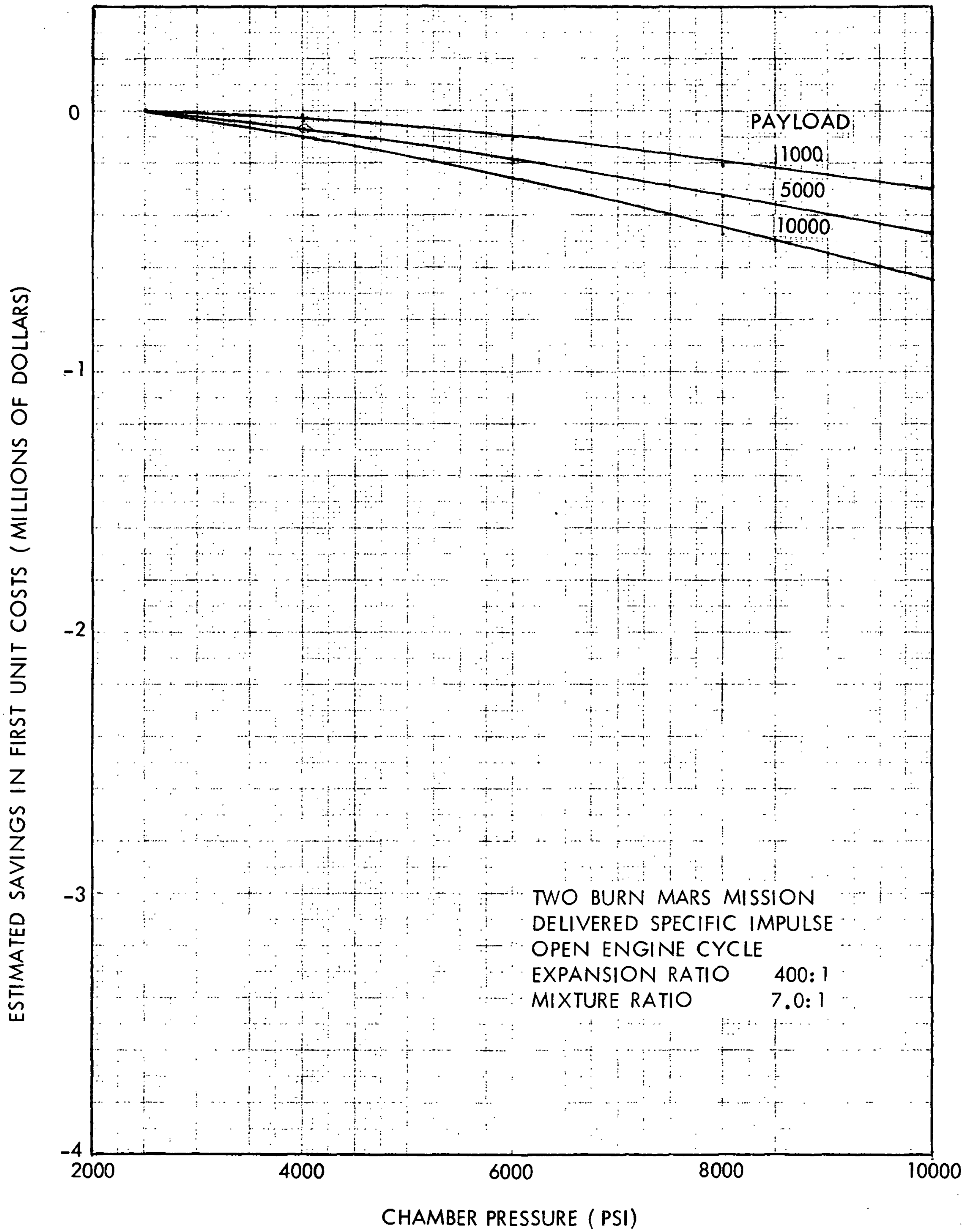

Figure 2-81. TFU Savings for Mars Mission Stage
(Delivered Open-Cycle Specific Impulse)

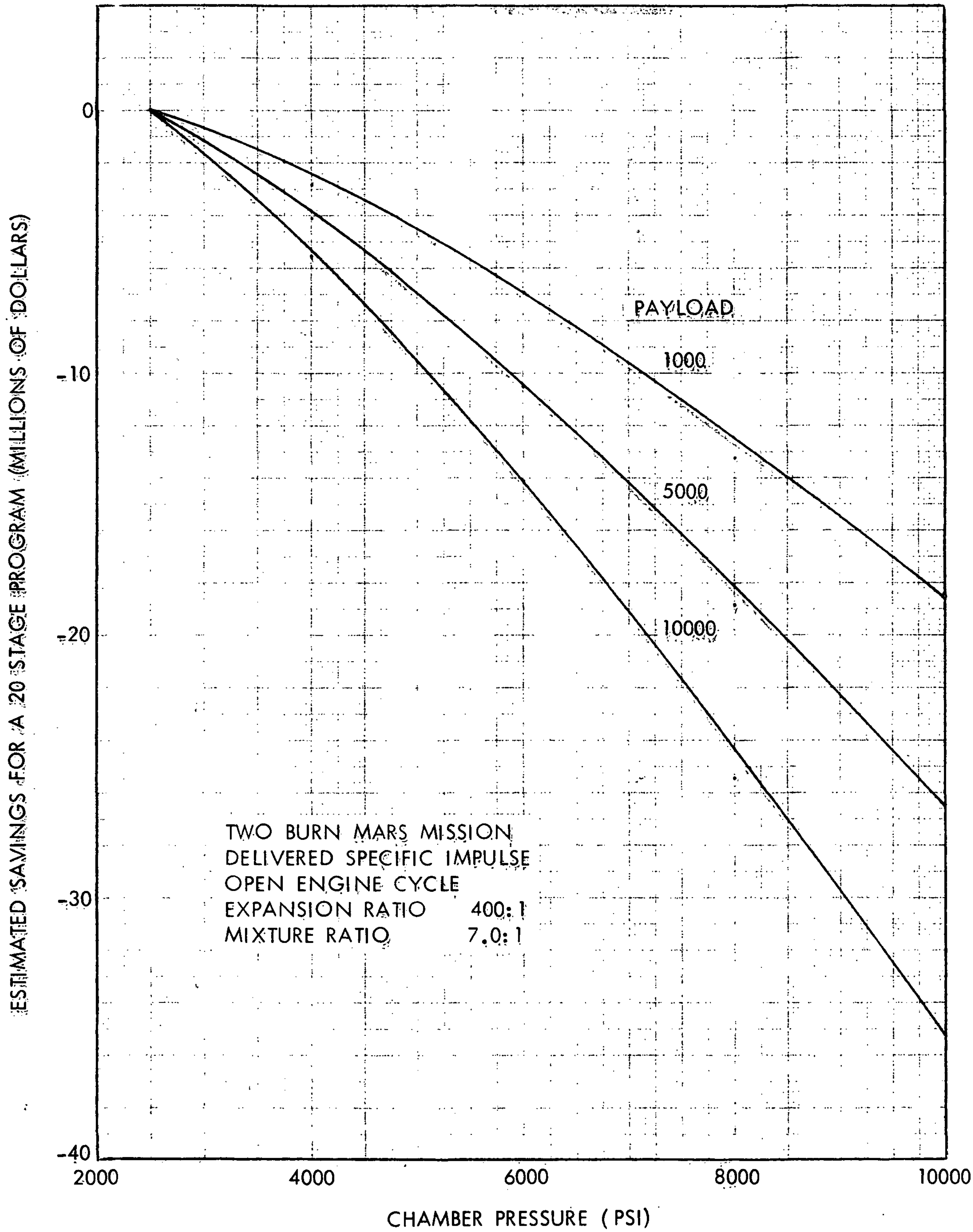

Figure 2-82. Program Savings for a Mars Mission Stage
(Delivered Open-Cycle Specific Impulse)

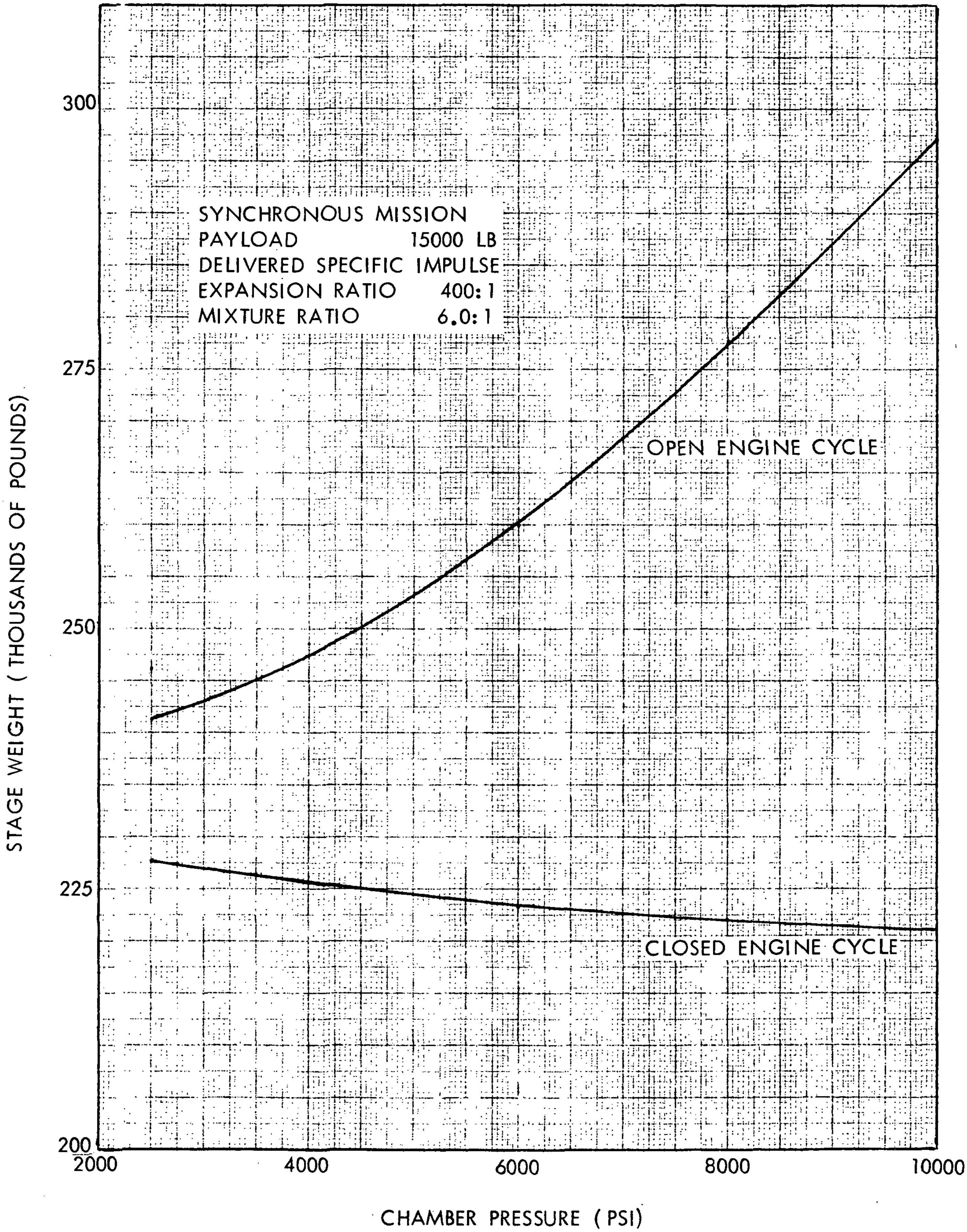

Figure 2-83. A Comparison of Stage Weights for a Synchronous Mission

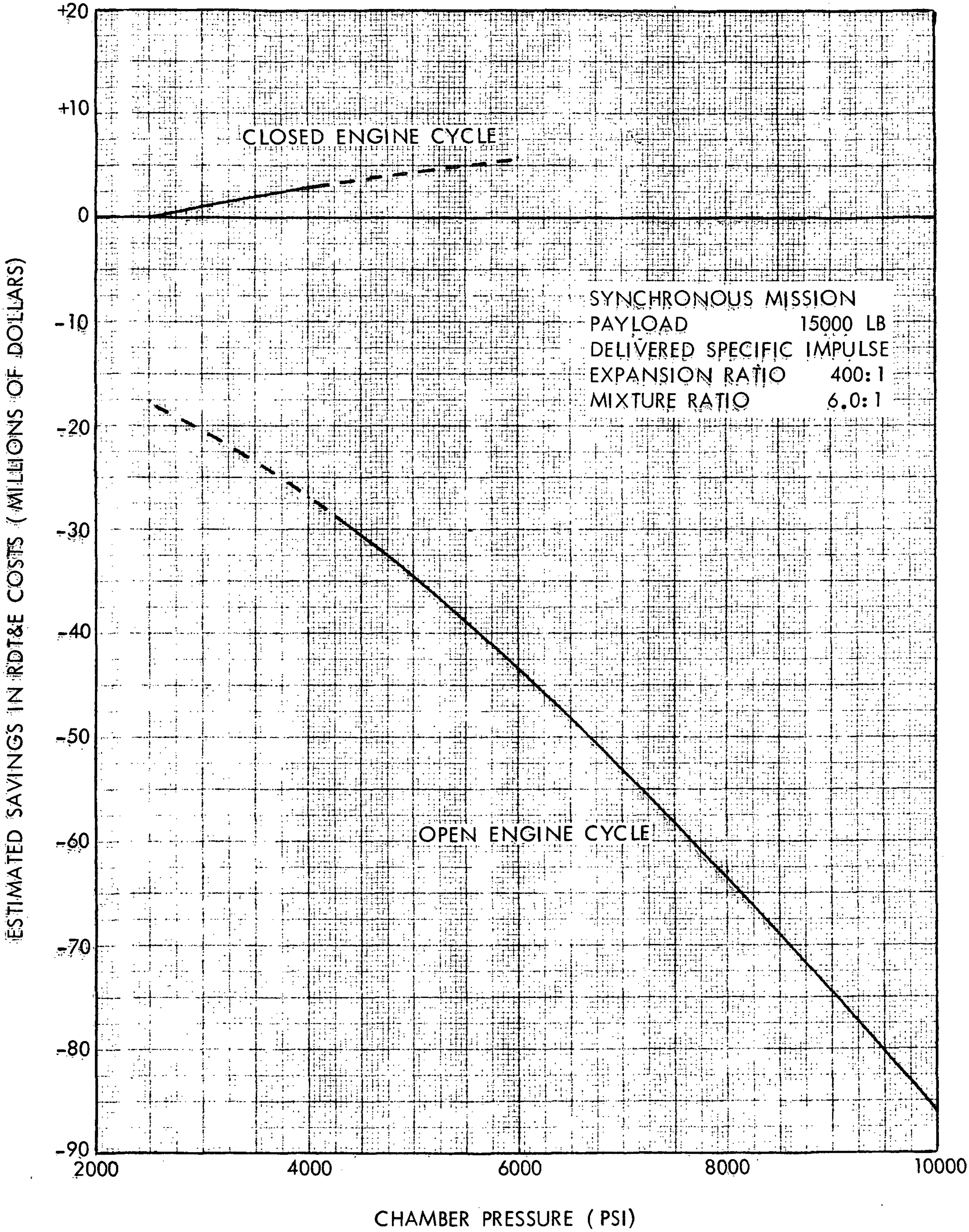

Figure 2-84. A Comparison of RDT&E Cost Differences for a Synchronous Mission Stage

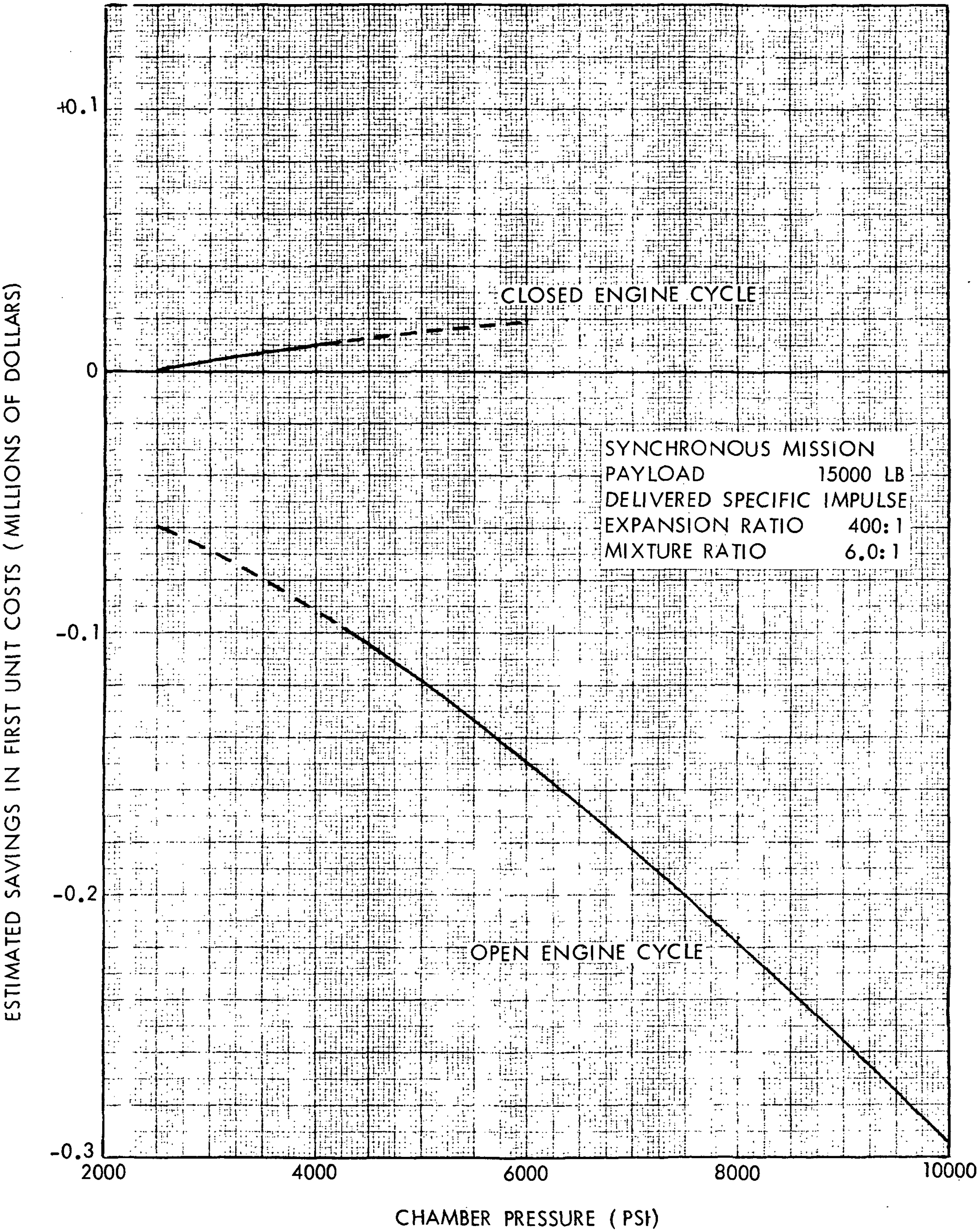

Figure 2-85. A Comparison of TFU Cost Differences for a Synchronous Mission Stage

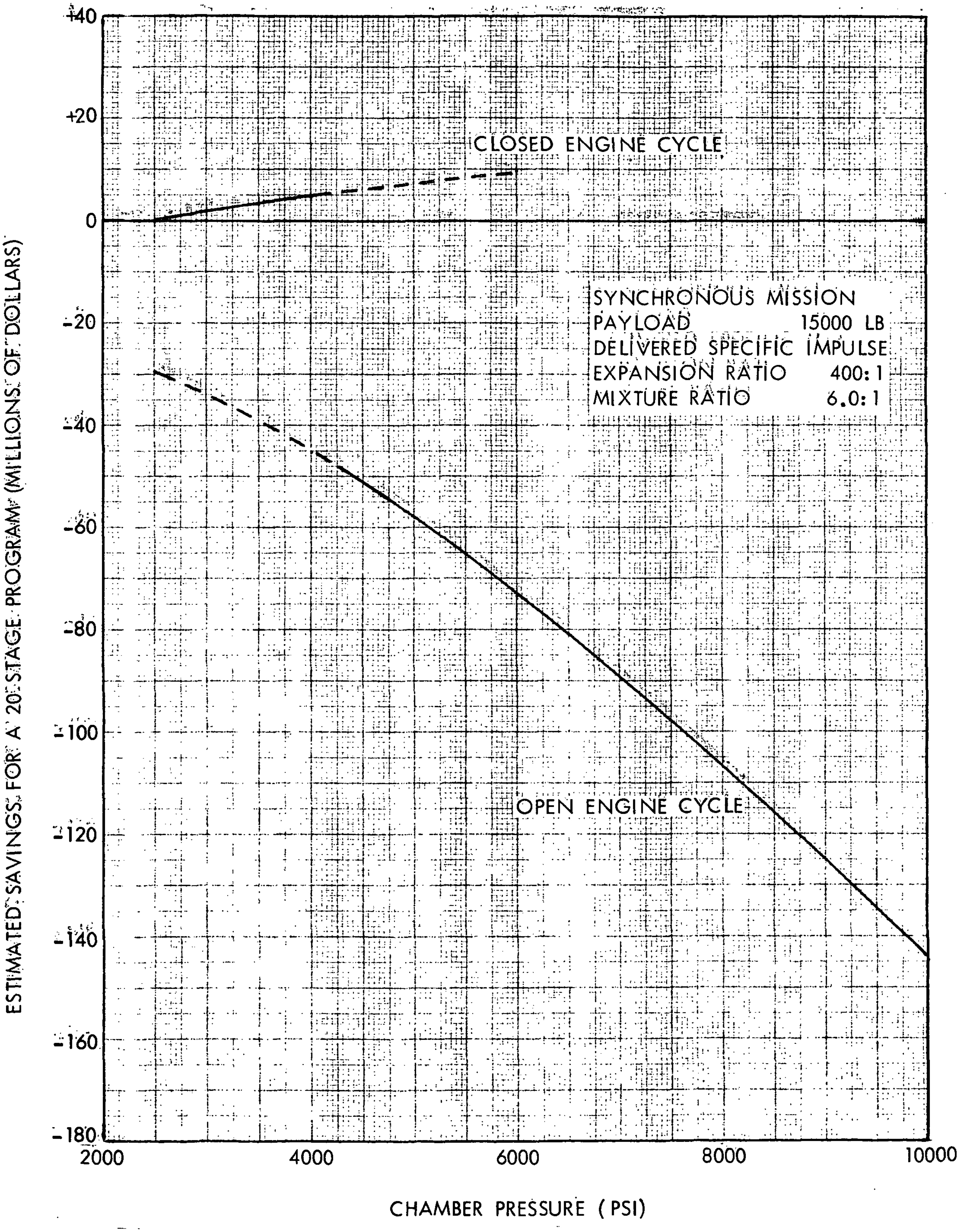

Figure 2-86. A Comparison of Program Cost Differences for a Synchronous Mission Stage

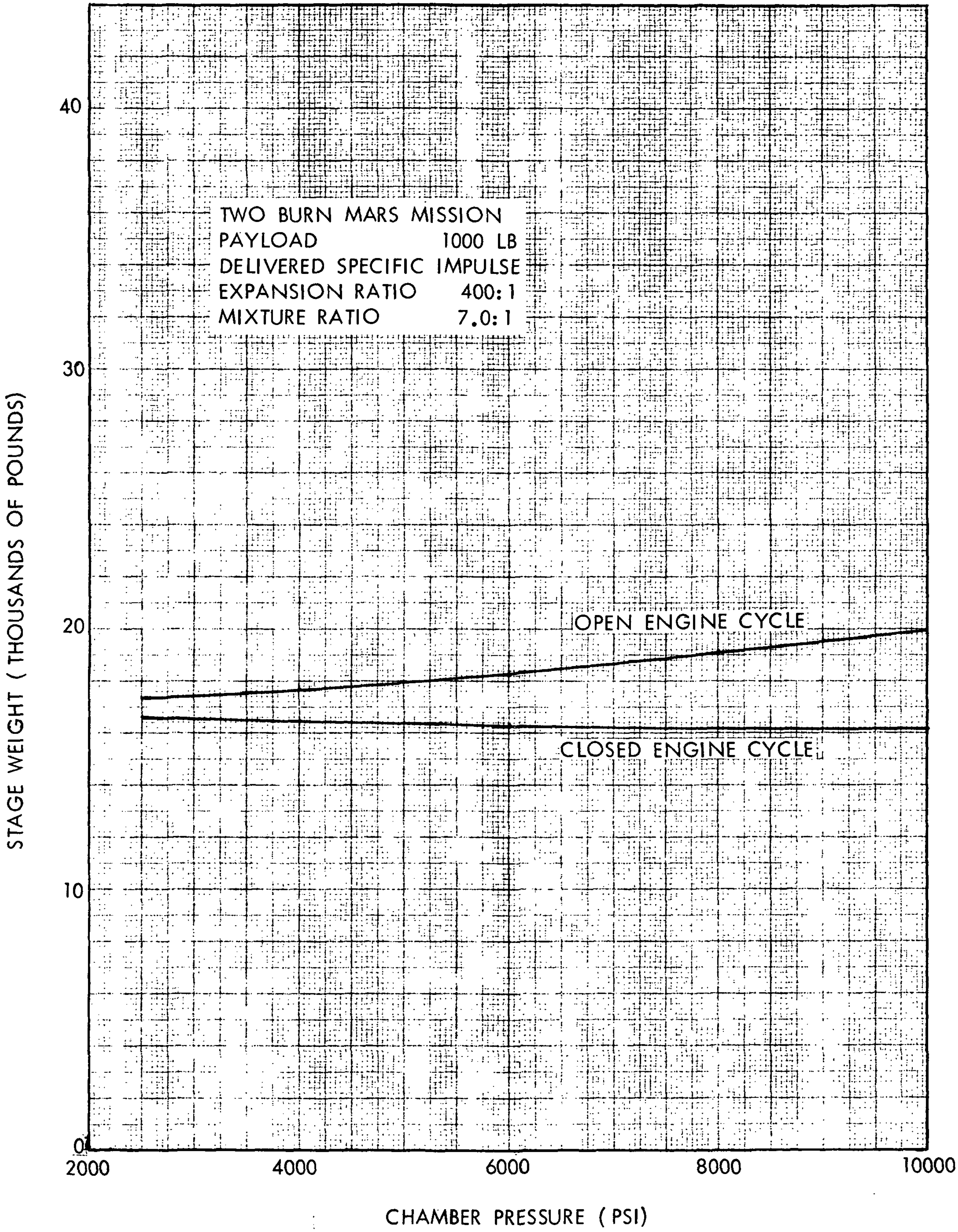

Figure 2-87. A Comparison of Stage Weights for a Mars Mission

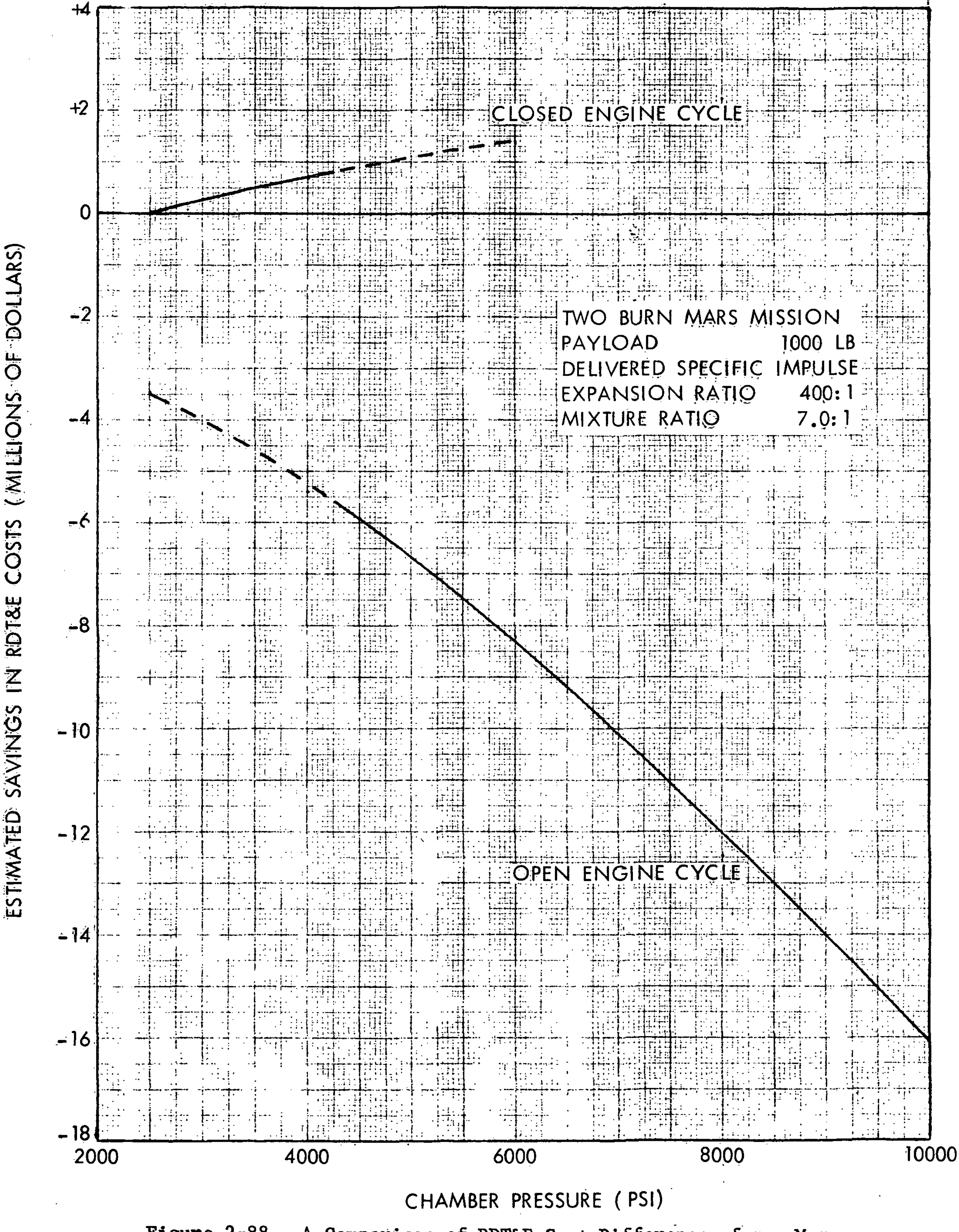

Figure 2-88. A Comparison of RDT&E Cost Differences for a Mars Mission Stage

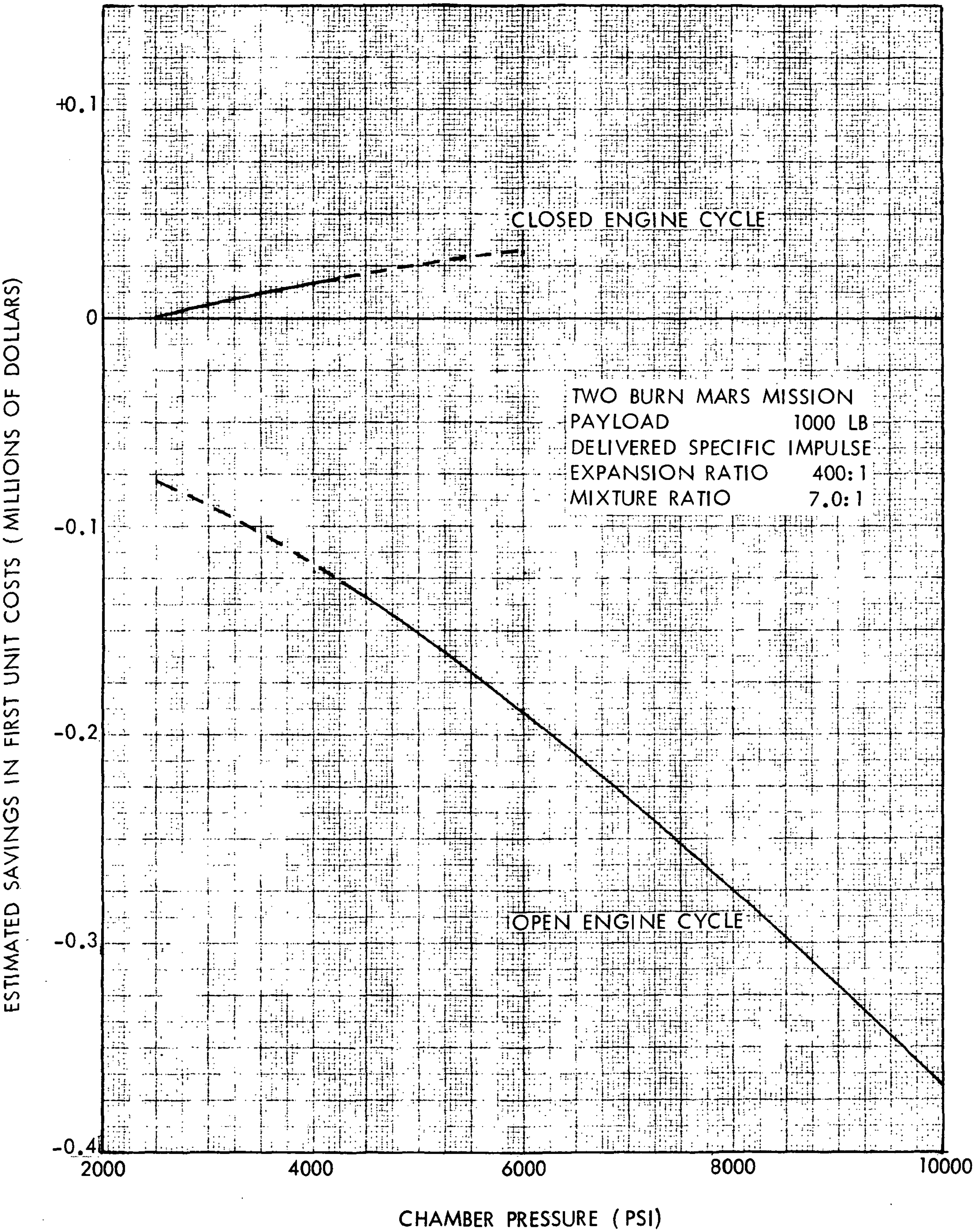

Figure 2-89. A Comparison of TFU Cost Differences for a Mars Mission Stage.

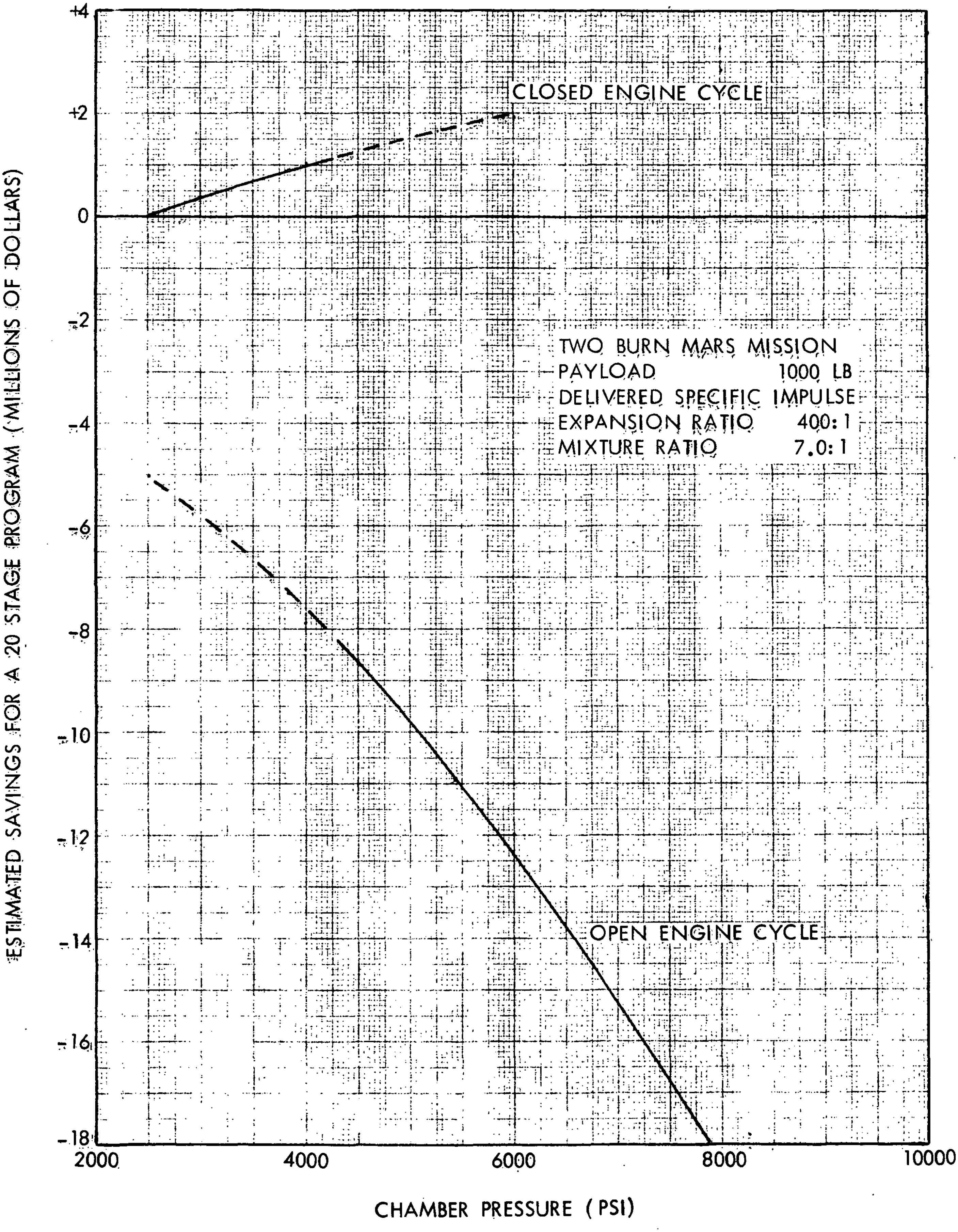

Figure 2-90. A Comparison of Program Cost Differences for a Mars Mission Stage

2.3.5 Conclusions Concerning Ultra High Chamber Pressures

This analysis indicates that, unless another engine cycle is developed, chamber pressures will be probably limited to less than 4000 psi for two reasons. The first being that the most efficient of engine cycles (closed) is theoretically limited to chamber pressures of about 6200 psi, and practically to pressures of 4000-4500 psi.

The second reason is that if there are no overriding considerations, such as additional performance is required to perform a necessary mission, then the high chamber pressures cannot be justified on the basis of cost savings. That is, the difference in the savings which would be realized during a program (five million dollars for 20 stages and less for fewer stages) and the cost required to develop the higher chamber pressure would not yield a larger enough return, if any, to base the need for higher chamber pressure on cost alone.

2.4 EFFECT OF MODERATE CHAMBER-PRESSURE INCREASES

As a result of the analysis of ultra-high chamber-pressure increases, a study was undertaken to assess the implications that decreases in stage length (due to moderate increases in chamber pressure) would have on program costs, including the shuttle transportation costs. This was accomplished by determining the number of shuttle launches required before the savings obtained from the smaller stages would equal the cost of developing the necessary chamber-pressure technology resulting in a smaller stage.

Three different synchronous mission stage sizes were considered for chamber pressures ranging from 500 to 3000 psi. The results obtained in this study are discussed in the subsequent portions of this subsection.

2.4.1 Data and Assumptions

Through the study, certain constraints, guidelines and pertinent design data were used. These are summarized in this section. Table 2-16 gives the design constraints used. Table 2-17 presents the prime structure data used in computing the shell and thrust structure weights. Table 2-18 summarizes the assumed tankage design data, including pertinent thermal and meteoroid protection data. The weights assumed for the astrionics systems and other miscellaneous subsystems are given in table 2-19. These weights reflect the miscellaneous subsystem philosophy recently being considered in an in-house MSFC Space Tug Study [7].

The stage geometry selected as the baseline for this analysis was the 40121 configuration, which has a large single hydrogen tank with ellipsoidal domes, and four small oxygen tanks with hemispherical bulkheads suspended below the thrust cone. This stage was selected instead of the tandem tank configuration because a shorter stage geometry will be more advantageous for use in the cargo bay of the shuttle. A typical stage geometry is shown in figure 2-91.

The parametric oxygen-hydrogen engine system performance, weight and geometry data used in this study were obtained from Rocketdyne for use in the "LOX/Hydrogen Engine Technology for Advanced Missions" study, contract NAS7-790. These data covered engines utilizing topping, expander and gas

Table 2-16. Summary of Stage Design Constraints

Constraint \ Mission	Single Stage Synchronous
Maximum Stage Diameter (In.)	174.0
Shell - Tank Spacing (In.)	6.0
Tank - Tank Spacing (In.)	6.0
Engine - Tank Spacing Factor (Chamber)	4.0
Engine - Tank Spacing Factor (Exit)	0.8
Engine - Booster Spacing (In.)	0.0
Engine Gimbal Angle (Degrees)	3.0
Thrust - To - Weight Ratio	0.25
Axial Acceleration (G's)	1.00
Lateral Acceleration (G's)	0.05
Payload Density (Lb/Ft^3)	25.0
Inert Weight Contingency Factor (%)	7.5

Table 2-17. Summary of Structural Design Data

Data \ Structure	Shell	Thrust Cone
Material	Aluminum	Aluminum
Density (lb/ft^3)	183.0	183.0
Material Strength (psi)		
Tension	67,000	67,000
Compression	46,000	46,000
Modulus of Elasticity (psi)	10^7	10^7
Safety Factors		
Tension	1.25	1.25
Compression	1.00	1.00
Monocoque-to-Complex Structure Weight Ratio	*	*
Spider Beam Multiplication Factor	N/A	N/A

* A function of diameter and limit load; see appendix C.

Table 2-18. Summary of Tankage Design Data

Data \ Mission	Synchronous
Tankage	
Material	Aluminum
Density (lb/ft^3)	183.0
Allowable Stress (psi)	60,000
Factor of Safety	1.10
Minimum Skin Gauge (In.)	0.025
Land Factors (Bulkheads)	0.10
Land Factors (Cylindrical Section)	0.05
Thermal Protection	
Initial Fuel/Oxidizer Temperature (°R)	36.0/162.6
Initial Fuel/Oxidizer Pressure (psi)	15.0/15.0
External Insulation Temperature (°R)	450.0/470.0
Insulation Density (lb/ft^3)	4.5
Insulation Thermal Conductivity (Btu/Hr-Ft-°R)	*
Meteoroid Protection	
Probability of no Punctures	0.995
Nominal Mission Altitude (n.m.)	200
Shield Material	Aluminum
Material Density (lb/ft^3)	183.0
Material Yield Stress (psi)	70,000
Minimum Skin Gauges (In.)	0.015
Miscellaneous	
Minimum Fuel/Oxidizer Ullage Volume (%)	5.0/5.0
Residual Fuel/Oxidizer Fraction (%)	2.0/2.0
Feedline Flow Velocity (fps)	20.0
Tank Support Factor	ξ

* A function of temperature and thickness; see appendix C.

ξ Dependent upon configuration; see appendix C.

Table 2-19. Miscellaneous Subsystem Weights (MSFC Tug)

Item	Weight (lb)
Electric Distribution	200
Electric Power Power Systems Fuel Cell Reactants	300
Communication/Instrumentation Communications Data Management Instrumentation	295
Guidance, Navigation, and Control Guidance, etc. Rendezvous and Docking Radar	210
Hydraulic/Pneumatic Purge Umbilical Tug/Orbiter Service	145
Propellant Utilization	35
Miscellaneous Destruct System Docking Adapter Subsystem Mounts Orbiter Interface Payload Interface Purge	1515
TOTAL	2700

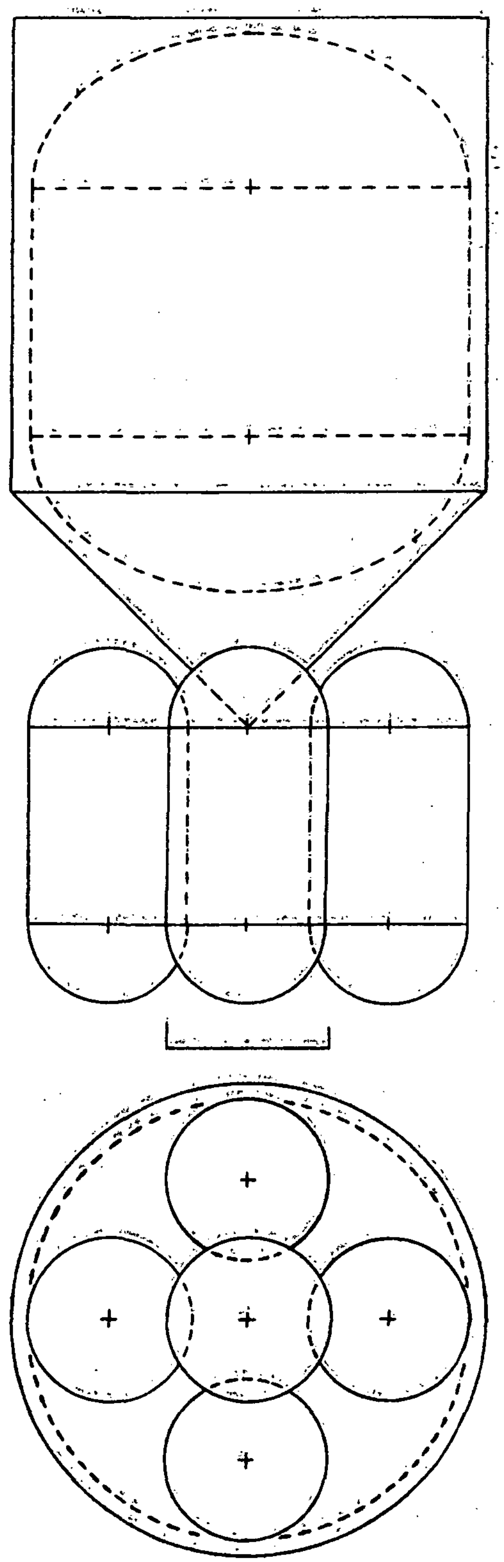

Figure 2-91. Multiple Oxidizer Tank Stage Configuration (40121)

generator cycles. The data for the topping and expander cycles included thrust levels from 15,000 to 120,000 pounds, area ratios from 100 to 400, and mixture ratios of 5.0, 6.0 and 7.0. Data were supplied for chamber pressures of 500 to 1000 psi, and 1000 to 3000 psi, for the expander and topping cycles, respectively. The gas generator cycle data covered lower thrust engines (5,000 - 15,000 lb) and chamber pressures of 800 and 1000 psi. These parametric engine data were published in appendix B of the final report [1] for the study cited above.

The costs generated during this analysis were based on cost estimating relationships which are predicated on historical cost data and pertinent vehicle parameters.[4] In general, the cost estimating relationships of any cost element contain coefficients which indicate the technology level and complexity of that individual element. Table 2-20 presents the technology level assumed for the main systems on the stage. Table 2-21 lists the percent learning curves used to compute the investment costs.

The program cost data developed during this analysis include only the RDT&E, investment, and upper stage propellant costs. The program costs presented do not contain the other cost elements normally included in the operations costs, because they are mainly launch vehicle and mission model oriented, and for identical missions and similarly sized upper stages the operational costs are relatively insensitive to variations in upper stage weight. Hence, any program cost sensitivities which are determined from these program cost data will be of sufficient accuracy.

2.4.2 Mission Profile

The mission profile selected for this analysis was the return of variously sized payloads from synchronous orbit. This mission profile, illustrated in figure 2-92, is similar to one being considered for a MSFC Space Tug. This two-burn mission would require the liquid oxygen/liquid hydrogen stage to be launched and placed in synchronous orbit by another stage or by itself with the use of external drop tanks. That is, the stage has a full (internal) propellant load at the beginning of the synchronous orbit coast. The first burn utilizing the "on-board" propellants is the retro maneuver associated with the return Hohmann transfer and plane change at synchronous orbit. The second and final burn circularizes the stage into a low earth orbit.

The velocities used for this mission assume a Hohmann transfer between an equatorial (0-degree inclination) synchronous orbit, and a 28½-degree inclination, circular 100-nautical-mile orbit. The velocities were not corrected to account for the effects of the stage's initial thrust-to-weight ratio (T/W = 0.25) and specific impulse, nor were the effects of orbital regression on the velocity requirements considered.

2.4.3 Stage Size and Cost

Stages were analyzed with engines having chamber pressures ranging from 500 to 3000 psi and engine nozzle expansion ratios of 100:1, 250:1 and 400:1. The results of the stage sizing analyses are presented in figures 2-93 through 2-97. The stage weights which correspond to various engine area ratios are depicted as a function of chamber pressure in figures 2-93, 2-94, and 2-95, for stages capable of carrying payloads of 1000, 20,000 and 50,000 pounds, respectively.

Table 2-20. Technology Level of Systems

Area	Technology Level & Technique *
Structures	
Shell	SOA - Aluminum Sheet Stringer
Thrust Structure	SOA - Aluminum Sheet Stringer
Tankage	SOA - Aluminum Monocoque
Meteoroid Shield	SAO - Aluminum Monocoque
Tank Supports	ADV - Composite
Propellant Feedlines	SOA - Aluminum
Propulsion	
Main Engines	New, advance, reuseable low Pc, LH_2/LOX
Reaction Control Thrusters	SOA - Monopropellant
Miscellaneous Subsystems	
Electrical Power and Distribution	Adaptation of existing hardware to a new unmanned, reuseable upper stage
Electrical Communication	
Instrumentation	
Guidance, Navigation and Control	
Hydraulic/Pneumatic	
Propellant Utilization	
Destruct	

* SOA - State-of-the-art Technology
ADV - Advance Technology

Table 2-21. Investment Learning Curves

System	Learning Curve
Structures	90%
Propulsion	95%
Miscellaneous Subsystems	90%
Assembly and Checkout	90%

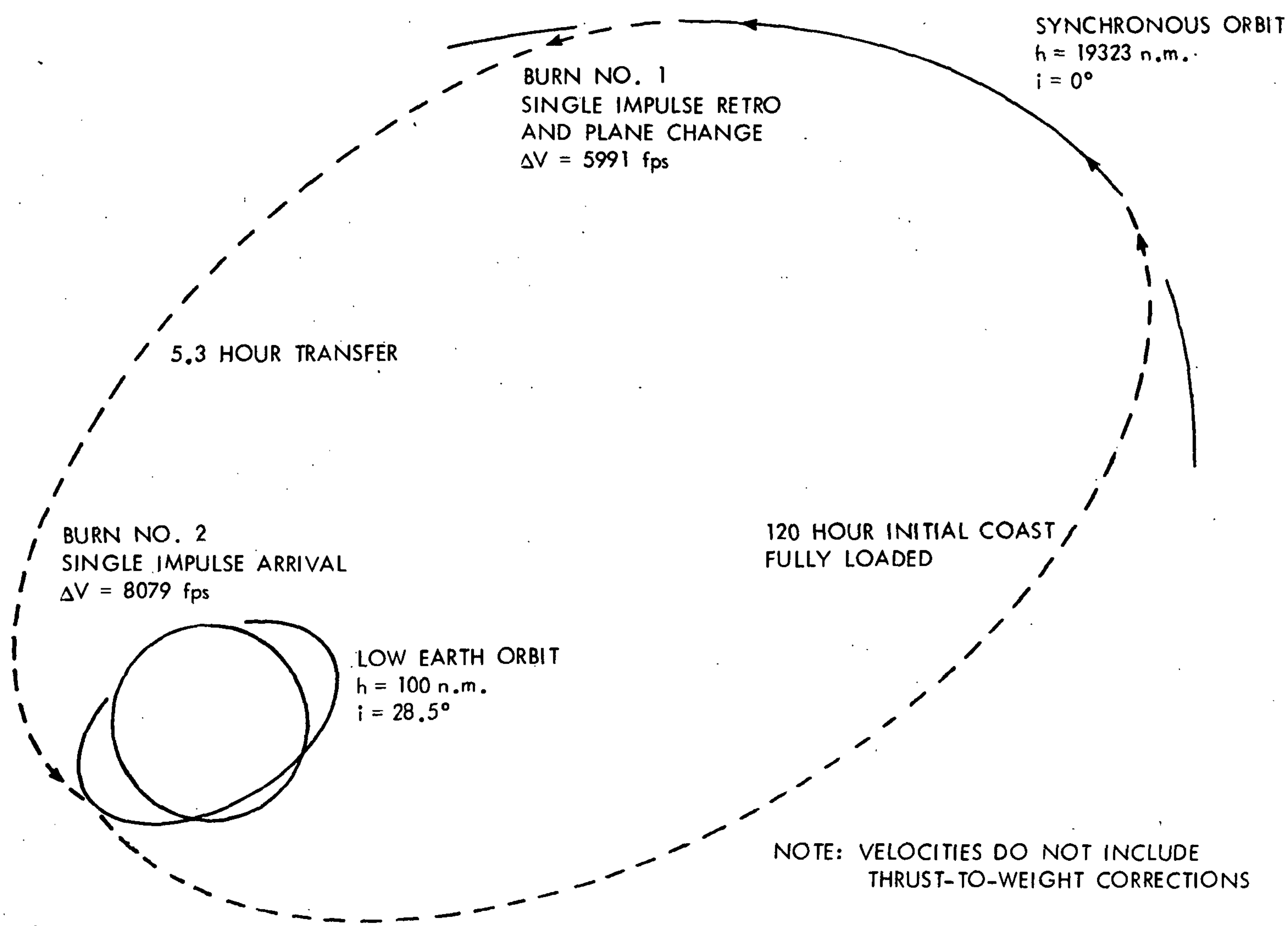

Figure 2-92. Mission Profile for Return Synchronous Mission

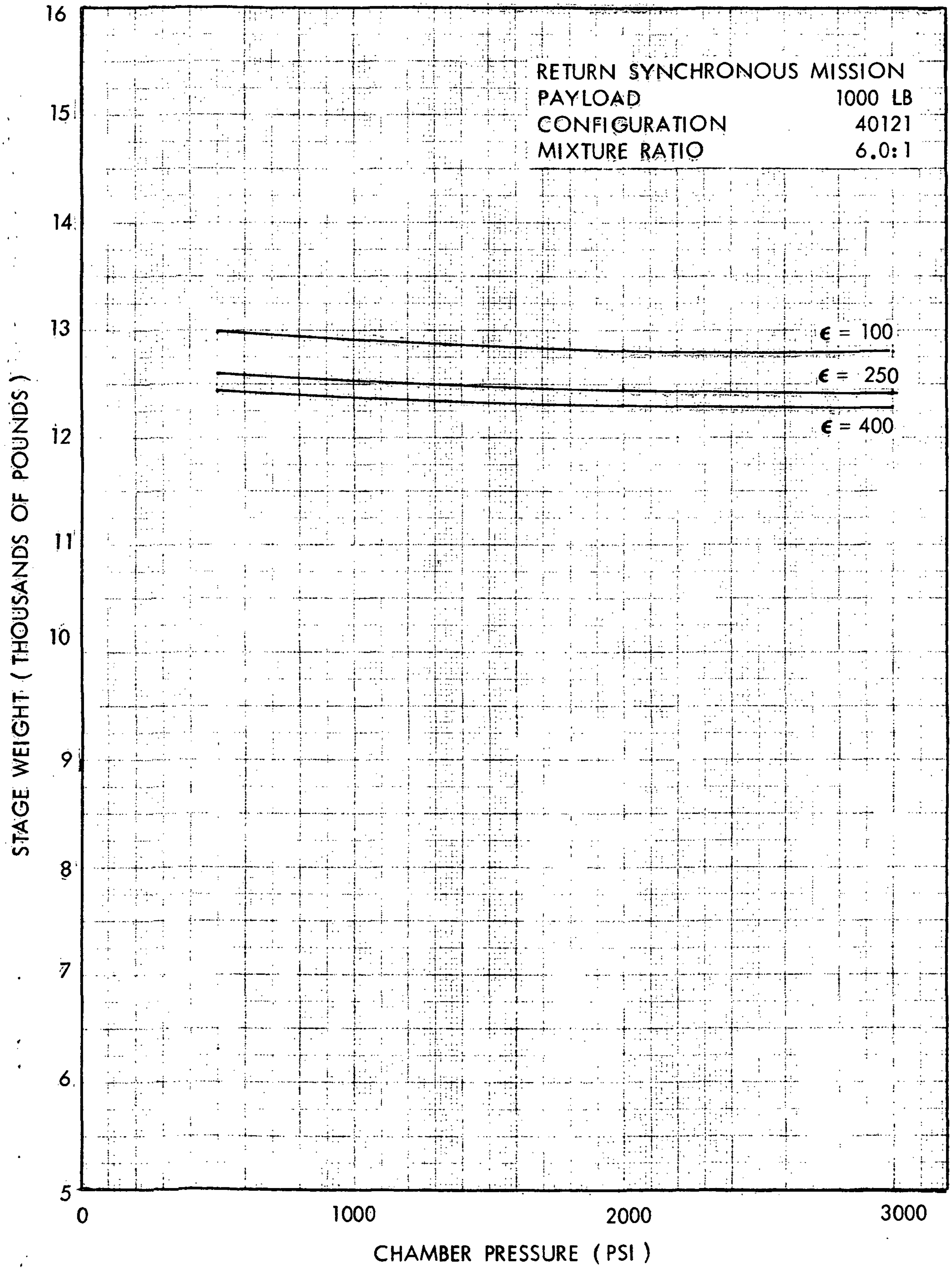

Figure 2-93. Stage Weights for a 1000-lb Payload

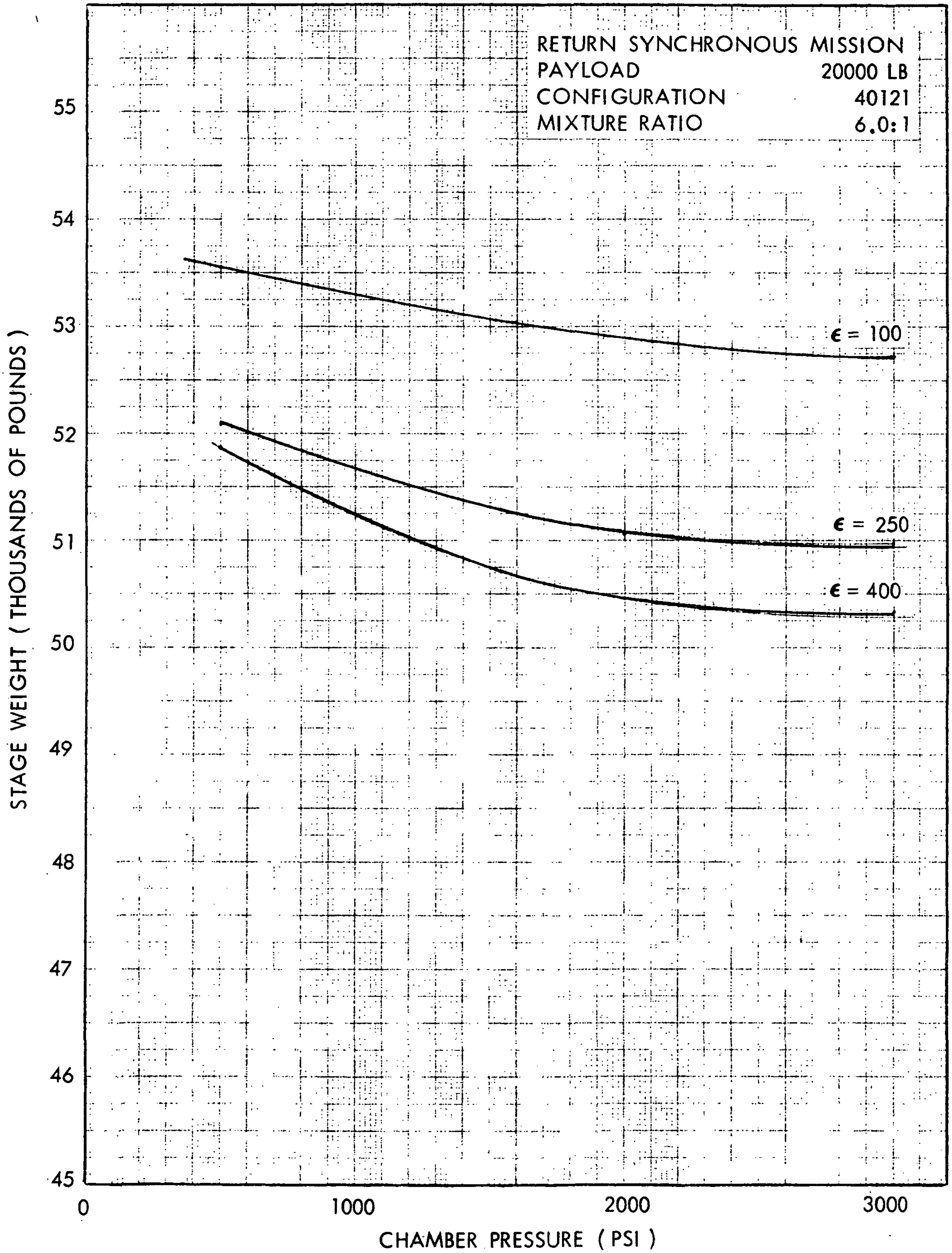

Figure 2-94. Stage Weights for a 20,000-lb Payload

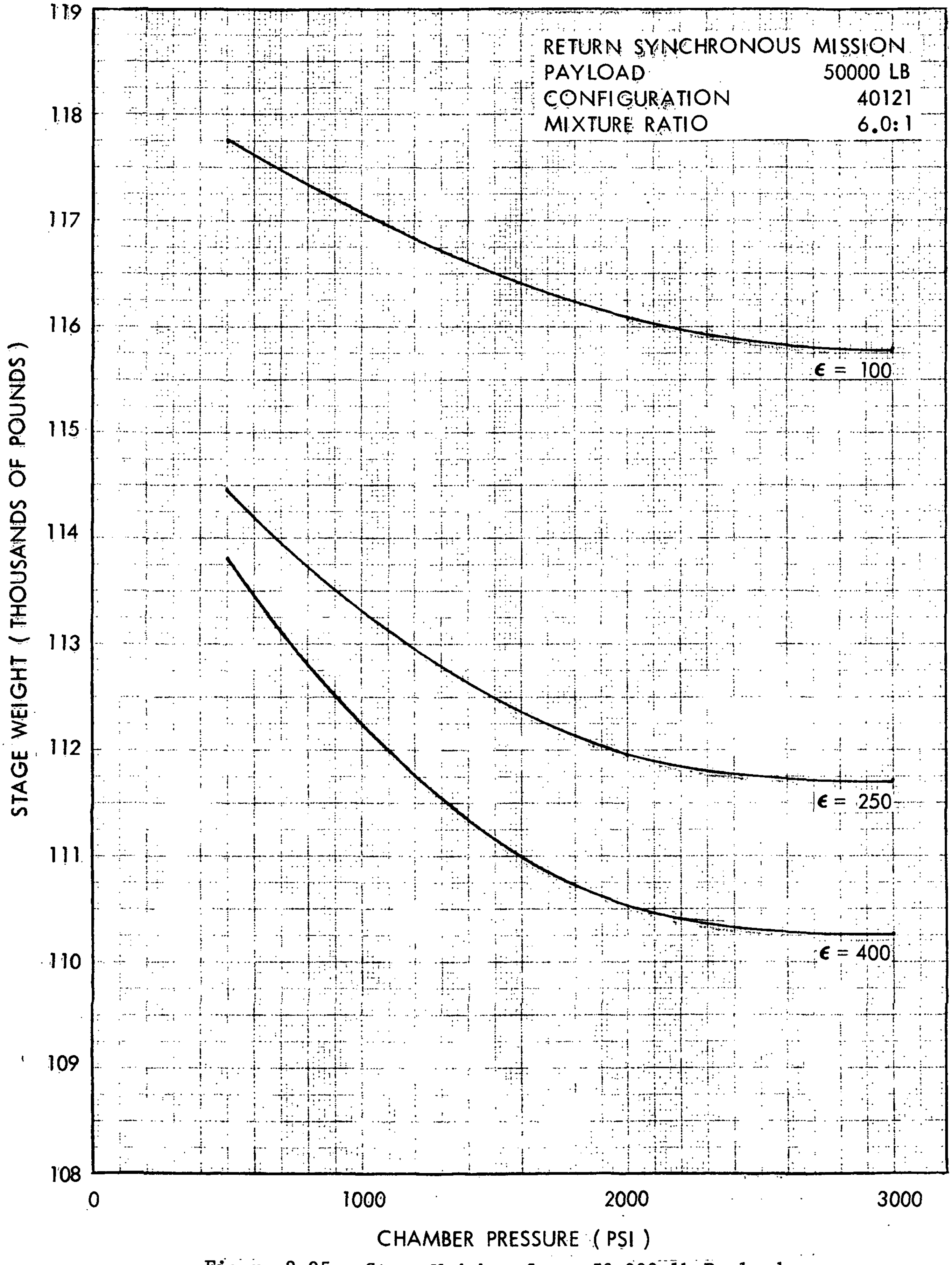

Figure 2-95. Stage Weights for a 50,000-lb Payload

The overall stage lengths are presented as a function of chamber pressure in figure 2-96.

The shuttle cargo bay volume required for the stages sized during this study are shown in figure 2-97. Although the maximum stage diameter varied from 123 inches (on the small 12-13K stage) to 174 inches (the maximum allowable diameter on the larger stages), the required shuttle cargo bay volume was computed on the basis of a useable cargo bay diameter of 174 inches. That is, there would be additional clearance, and hence unused space around stages whose diameters are less than 174 inches.

The RDT&E, TFU and program costs were determined for the various stages sized in this analysis. The costs were computed with a computer routine, developed specifically for upper stages, which determines costs from cost estimating relationships.(4) The resulting RDT&E, TFU and program costs are depicted for various stage sizes in figures 2-98, 2-99 and 2-100, respectively.

The 20-stage program costs shown in figure 2-100, include only the RDT&E, investment, and upper stage propellant costs. Other costs which are normally included in operations have not been included (see subsection 2.4.1).

2.4.4 Variations in Stage Size and Cost

The differences in stage size and cost were determined for the various stages investigated, using the stages with engines having 500-psi chamber pressures as references. The results, depicted in figures 2-101 through 2-106, show the reductions in stage size and cost which accompany increases in engine chamber pressure.

The reductions in stage weight, stage length and shuttle cargo volume required for the stage, which result from moderate increases in chamber pressure, are given in figures 2-101, 2-102 and 2-103, respectively. These data are presented for three differently sized stages and two area ratios. The payloads which correspond to the stage weights of 12-13,000, 51-55,000 and 109-119,000 pounds, are 1000, 20,000 and 50,000 pounds, respectively.

Figures 2-104, 2-105 and 2-106, show the estimated RDT&E, TFU and program cost savings obtained through increased chamber pressure. These data are presented for the two smaller sized stages and two nozzle expansion ratios. These cost data do not reflect the cost associated with developing the higher chamber pressure technology.

2.4.5 Technology Break-even Points

Because the costs of obtaining the higher chamber pressure technology were unknown, it was necessary to "back-into" the problem in order to ascertain the technology break-even points. These were found by determining the technology cost which would equal the cost saving resulting from a smaller, higher performance stage. That is,

Break-even Investment Costs = Stage RDT&E Savings

+ (Number of flights) (Investment Savings/Stage + Operations

Savings/Stage + Launch Savings/Launch).

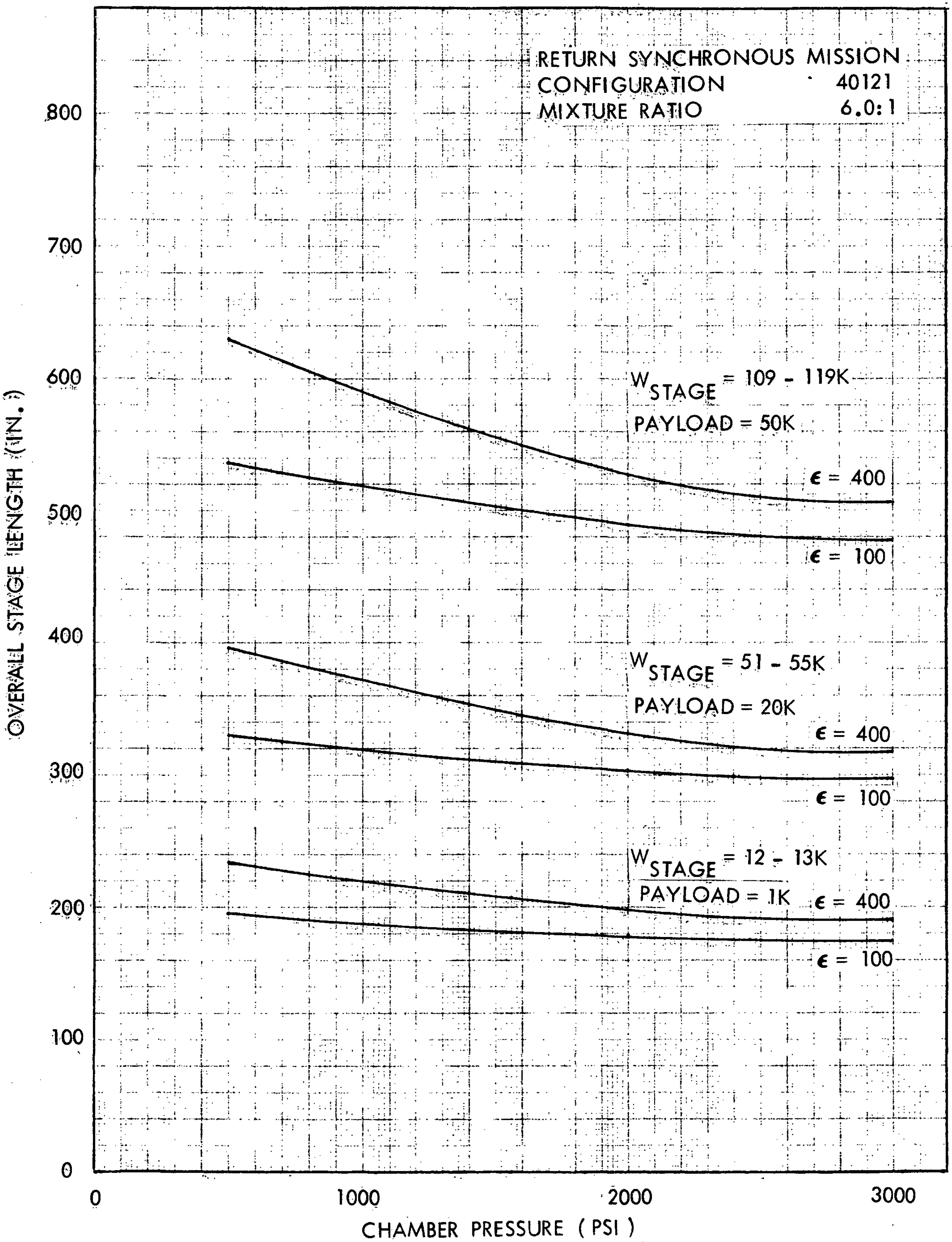

Figure 2-96. Stage Length of Variously Sized Stages

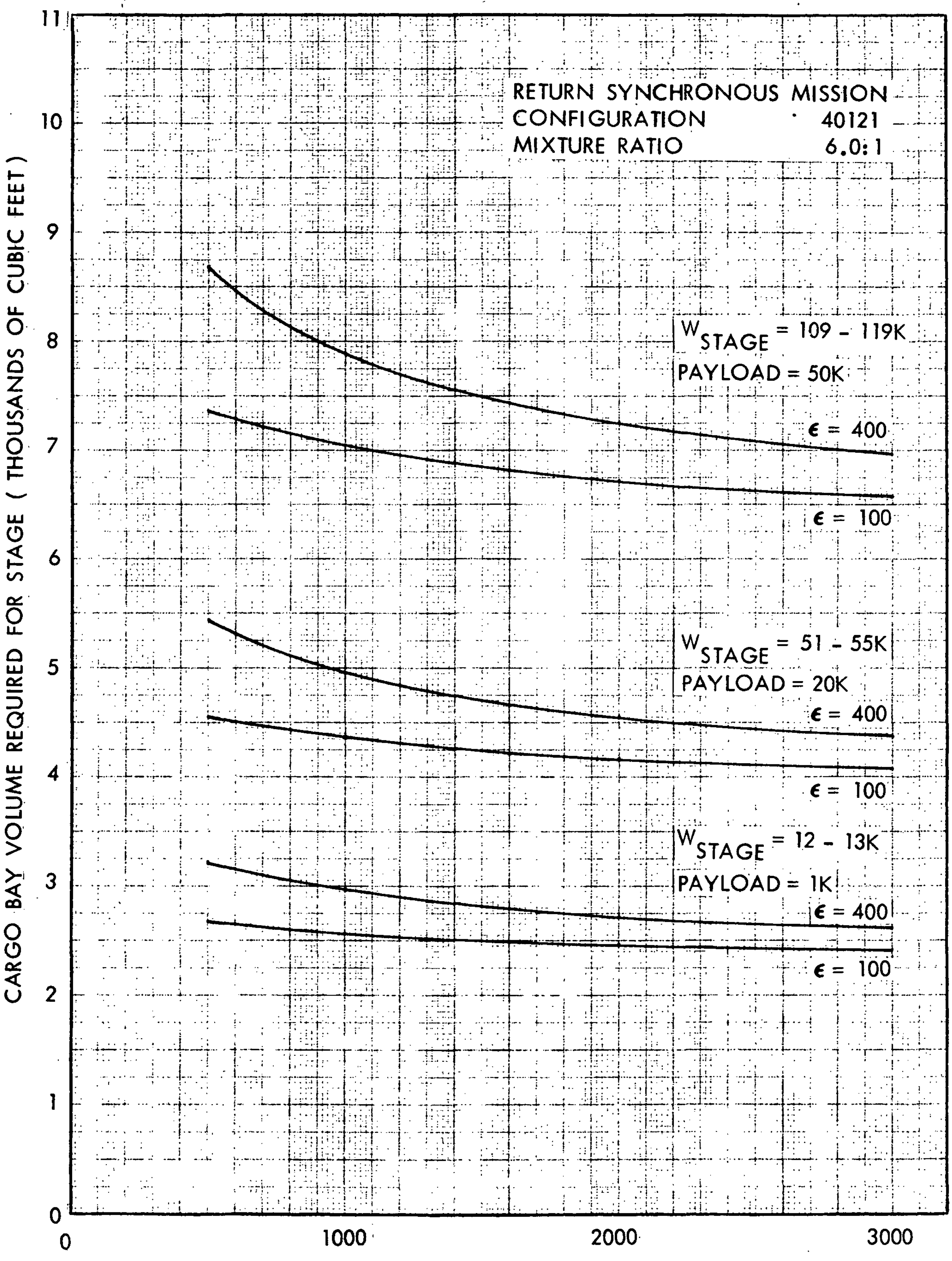

Figure 2-97. Cargo Bay Volume Required for Variously Sized Stages

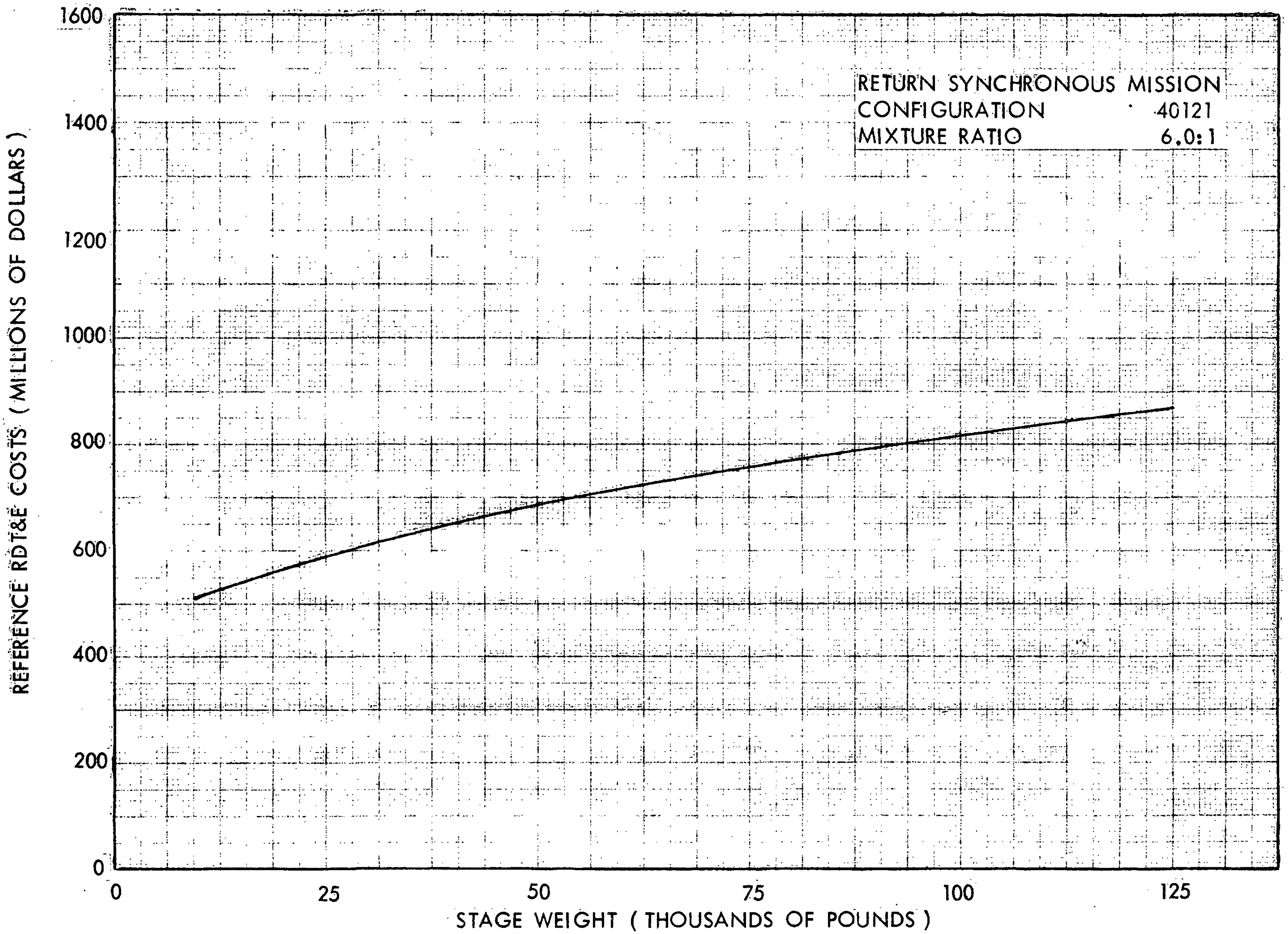

Figure 2-98. Reference RDT&E Costs

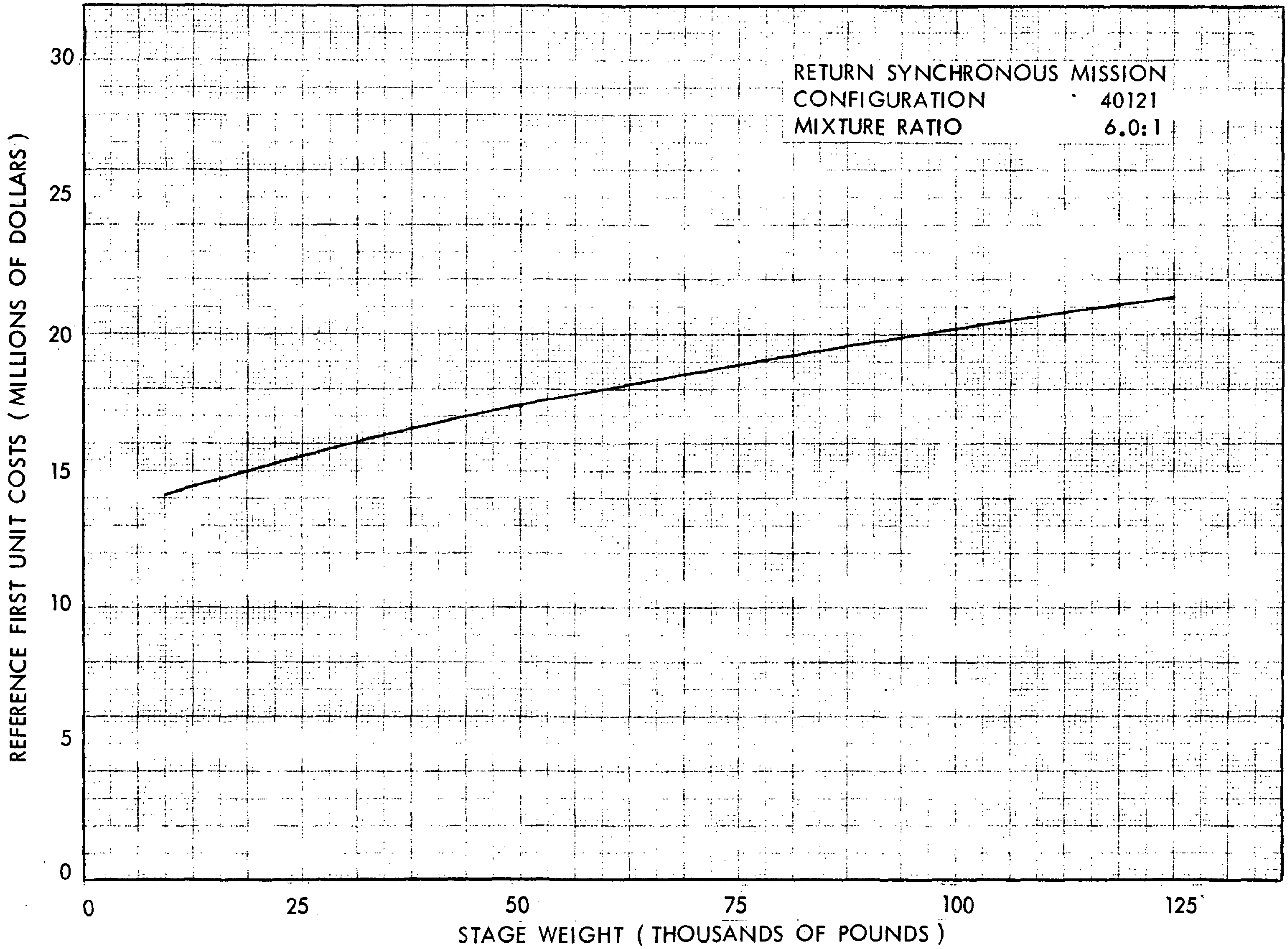

Figure 2-99. Reference TFU Costs

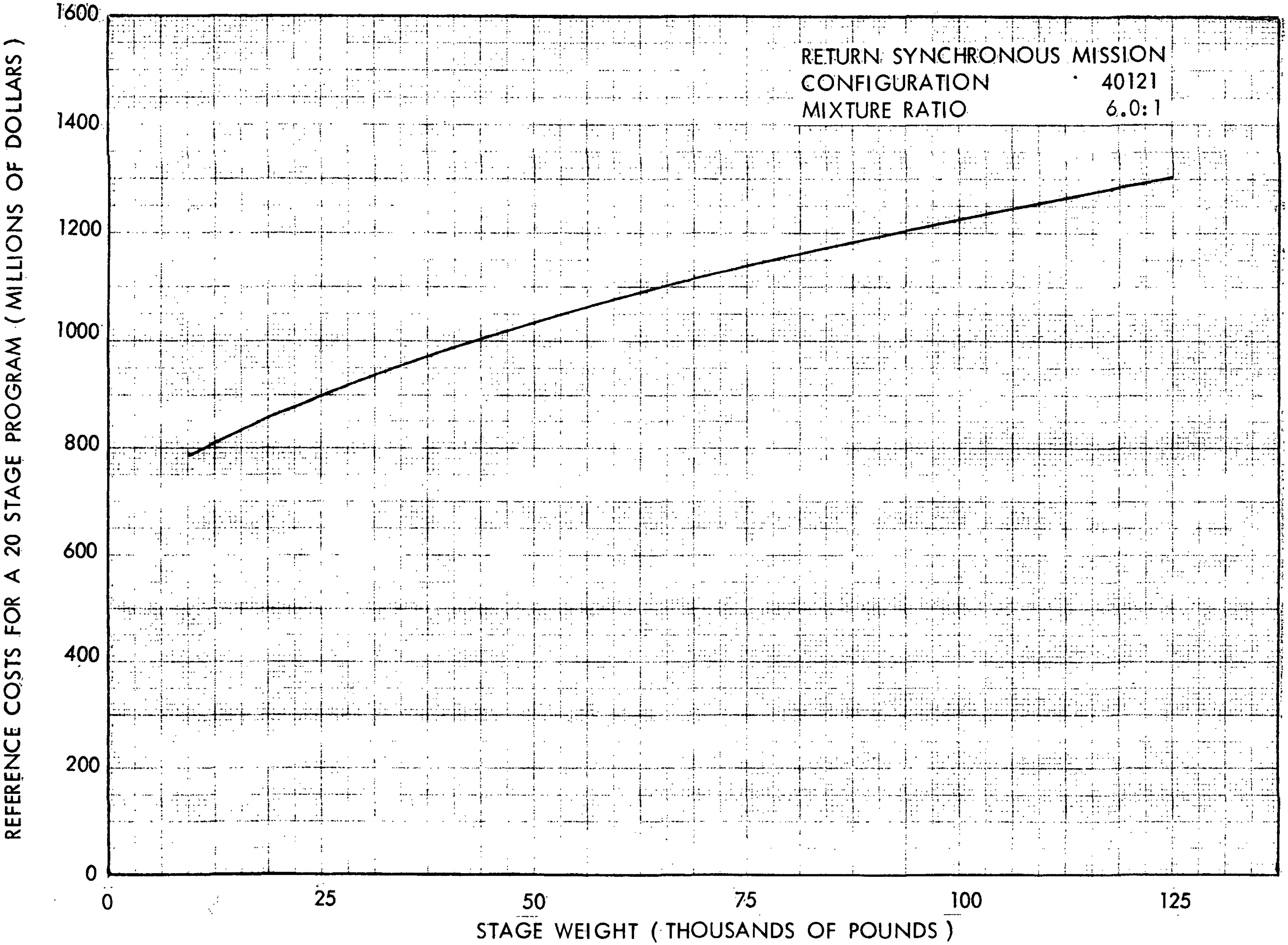

Figure 2-100. Reference Costs for a 20-Stage Program

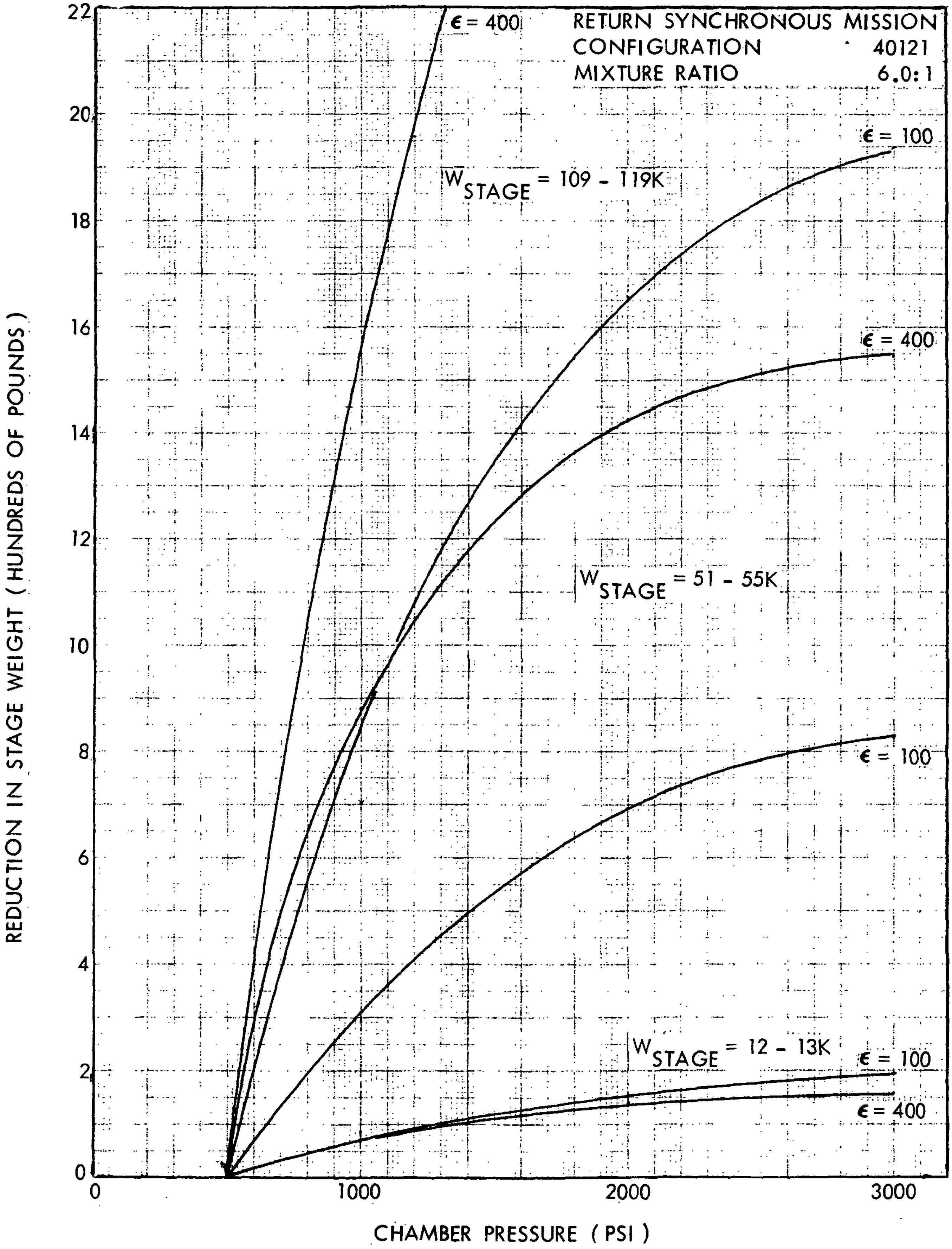

Figure 2-101. Reduction in Stage Weight

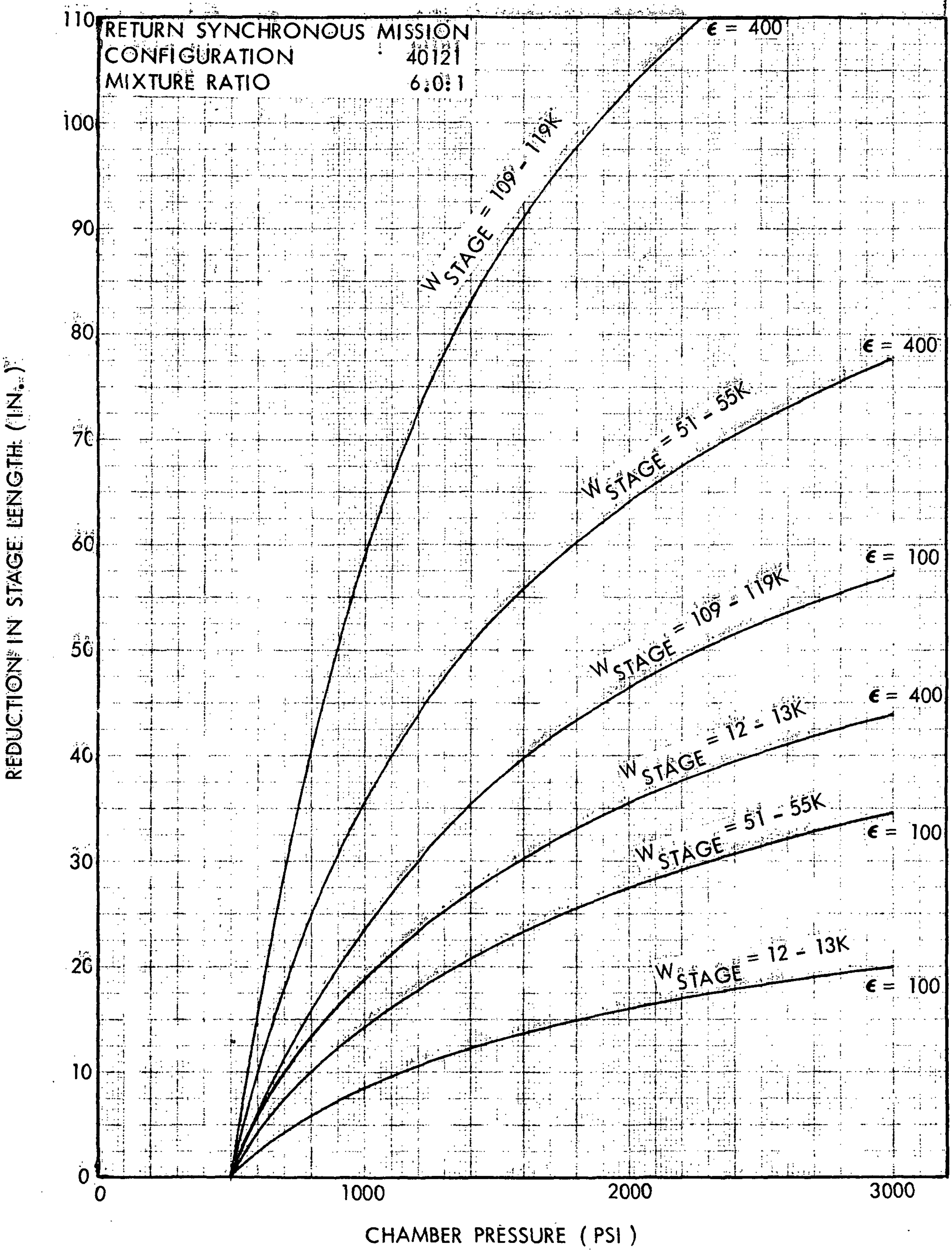

Figure 2-102. Reduction in Stage Length

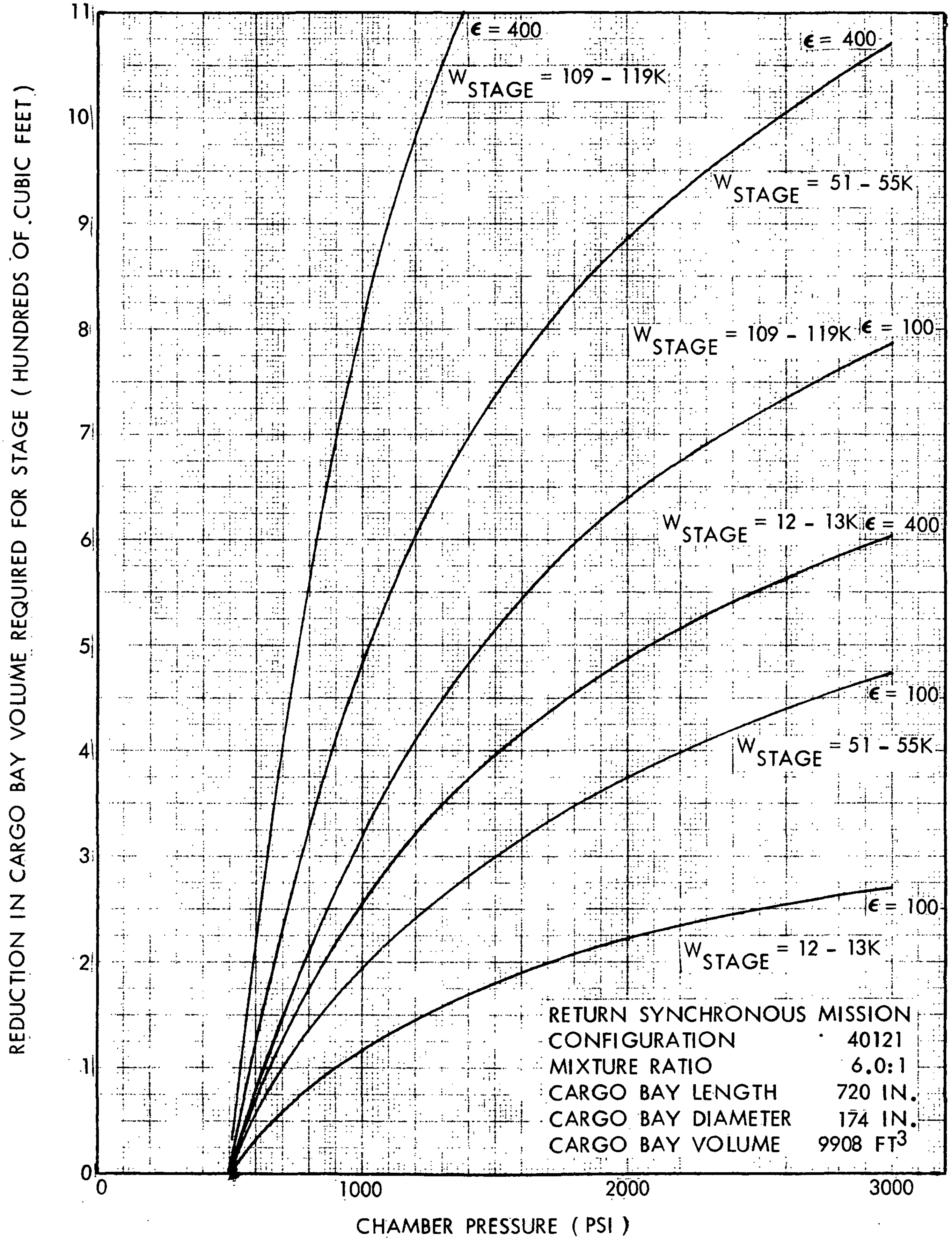

Figure 2-103. Reduction in Cargo Bay Volume Required for Stage

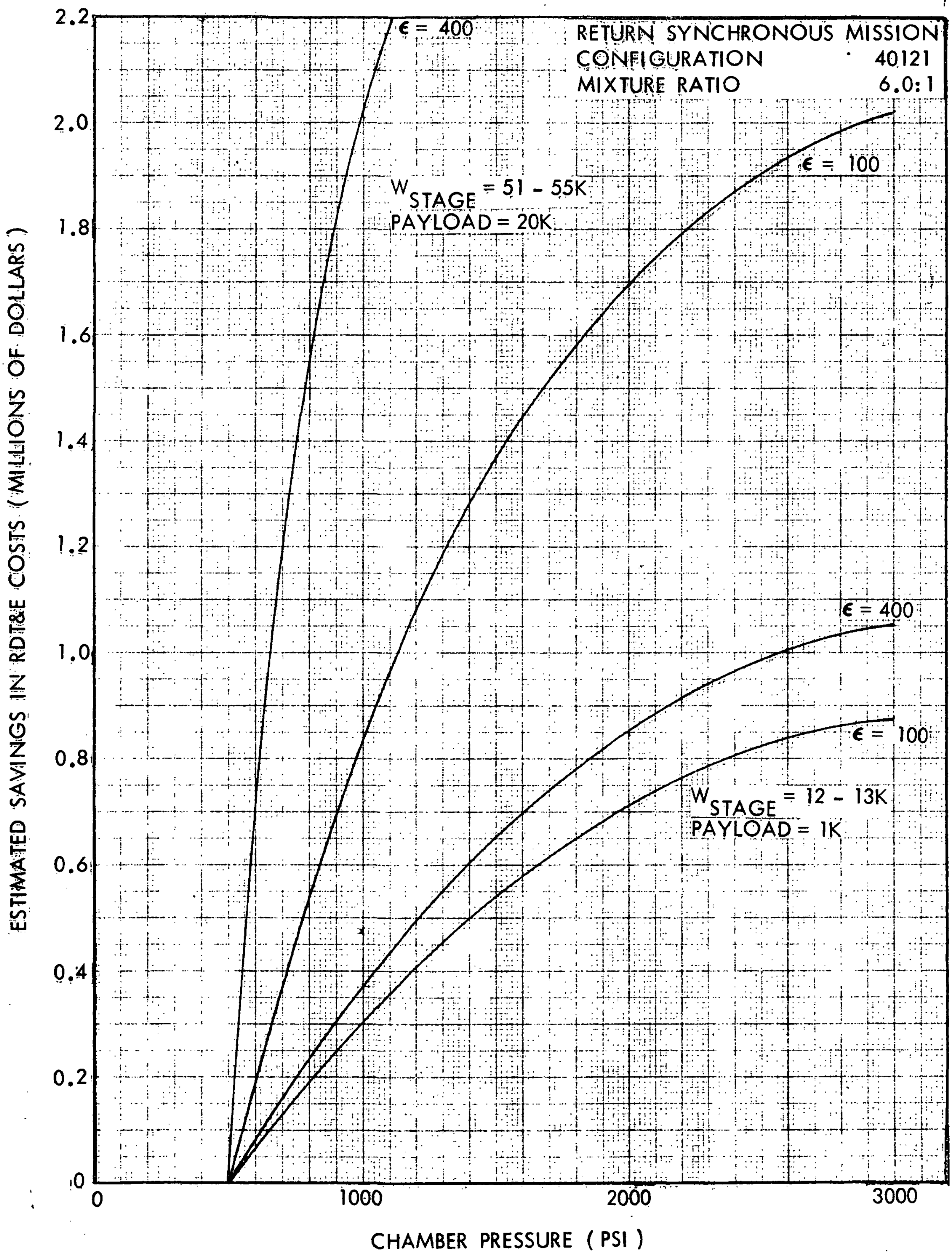

Figure 2-104. Estimated Savings in RDT&E Costs

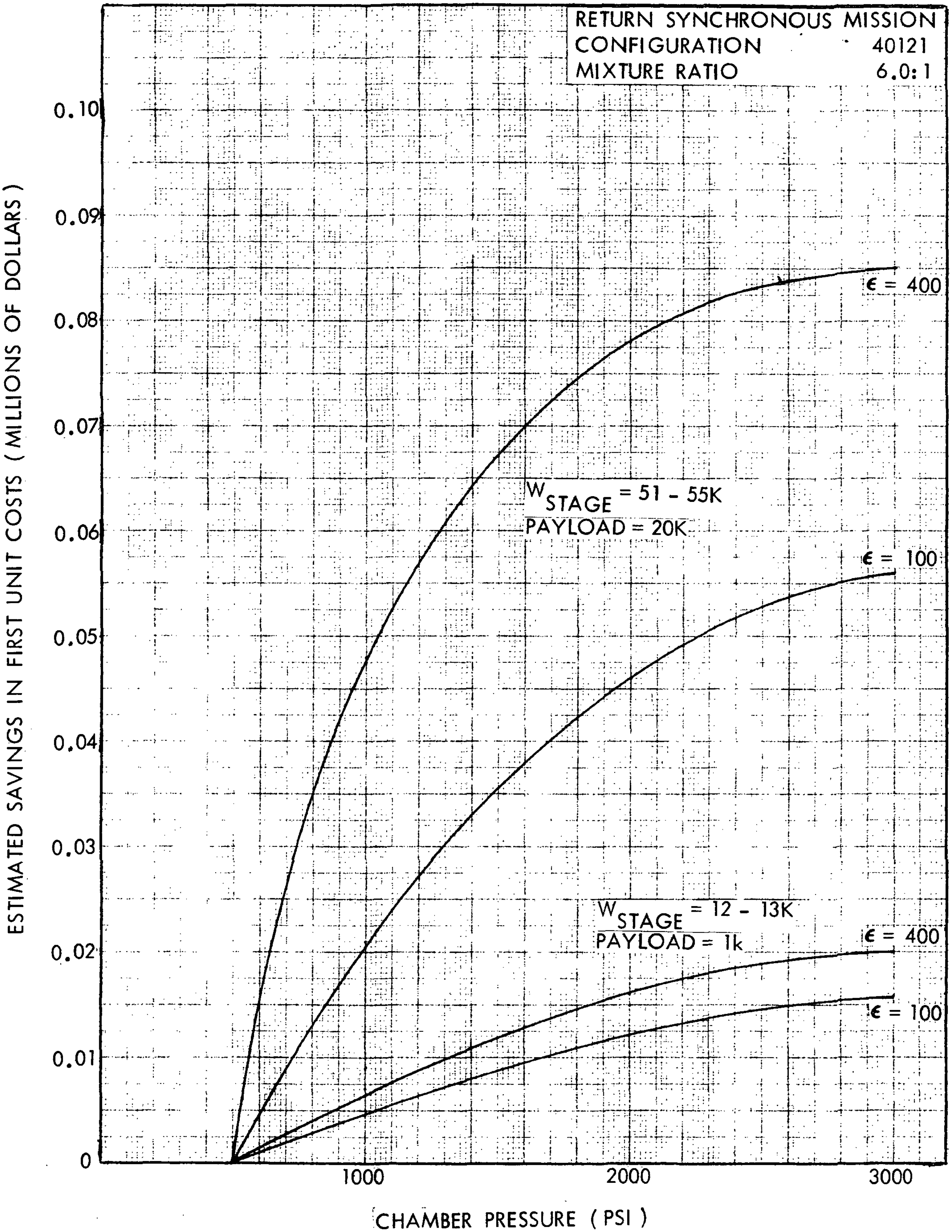

Figure 2-105. Estimated Saving in TFU Cost

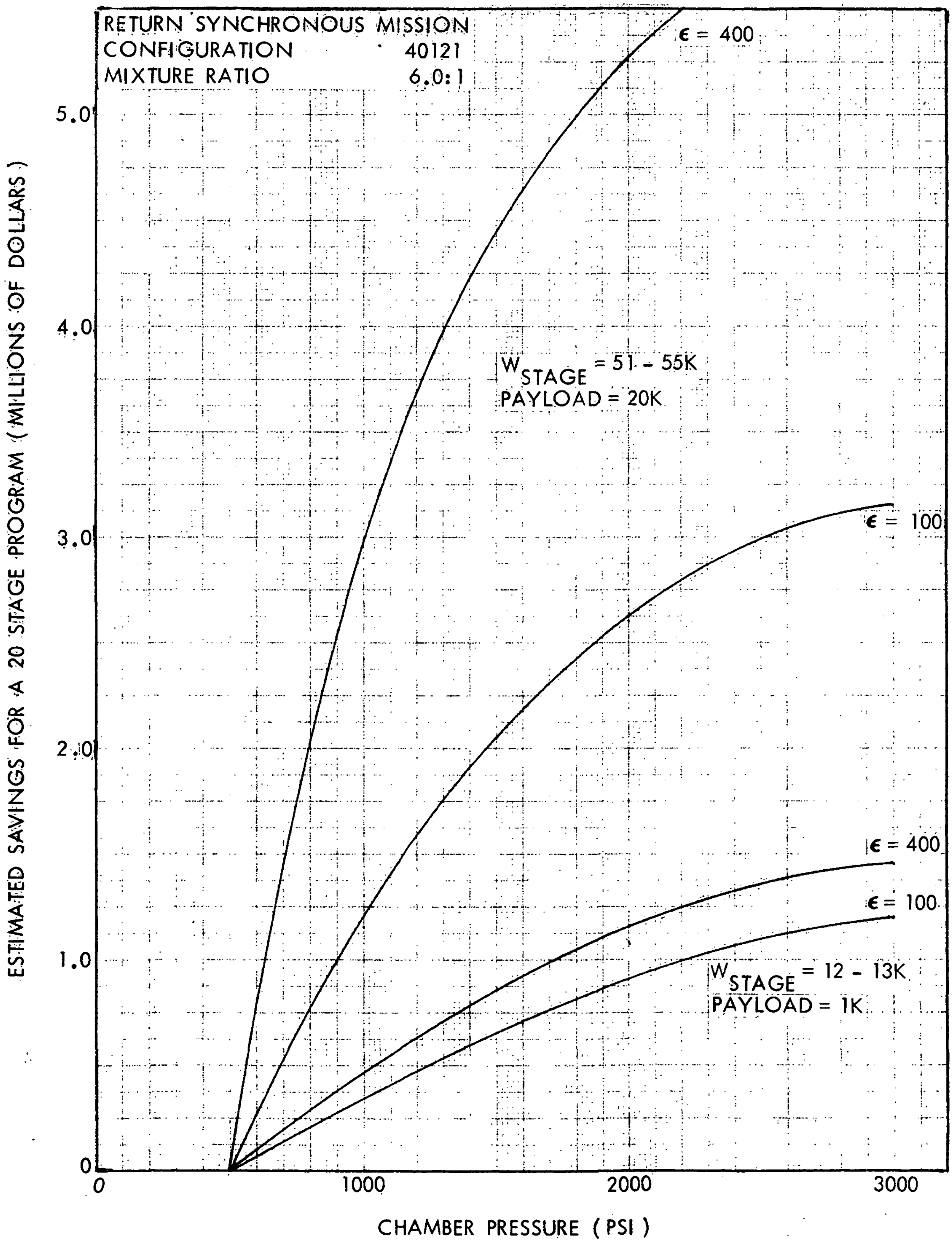

Figure 2-106. Estimated Savings for a 20-Stage Program

Although the RDT&E savings are independent of the program size, the other savings terms in parentheses are functions of the number of stages used in the program.

The savings associated with the stage RDT&E and program(investment and operations) can be readily found from the savings data presented in figures 2-104 through 2-106, and similar program cost data developed, but not presented, for programs having various numbers of stages. However, for the launch cost savings, it was necessary to assess the implications that the smaller stages (due to increased chamber pressure) have on shuttle launch costs.

This was accomplished by determining the number of shuttle launches needed before the additional extra payload gained in each launch of a stage (because of lighter, shorter stages) would equal the equivalent payload of a single shuttle launch without a stage. The extra cargo bay volume which can be utilized for other additional payloads is equal to the reduction in cargo bay volume required for the stage (see figure 2-103). The shuttle cargo bay criteria used in this analysis are presented in table 2-22. Various densities were assumed for the additional payload, and the computed extra payloads were checked to ensure that the volume and weight limitations of the cargo bay were satisfied.

Table 2-22. Shuttle Cargo Bay Criteria

Cargo Bay Length	60.0 ft
Cargo Bay Diameter	15.0 ft
Maximum Allowable Cargo Diameter	14.5 ft
Maximum Allowable Cargo Volume	9908 ft^3
Maximum Allowable Cargo Weight	65000 lb

The results of this analysis, which show the number of stage launches required to save one shuttle flight, are presented in figures 2-107 and 2-108, for two different stage sizes. Only the volume-limited cargo space case is presented in figure 2-107, for the smaller stage, because the critical density of the extra payload (the ratio of maximum allowable extra payload weight to volume) is unrealistically high for the weight limited case. Although the opposite is true for the larger stage (51-55,000 lb), similar data are presented for the larger stage in figure 2-108, for both the weight- and volume-limited cases.

The launch cost savings were computed from the number of stage launches required to save one shuttle flight, and used in conjunction with the stage savings data (figures 2-104 through 2-106) to determine the technology break-even points.

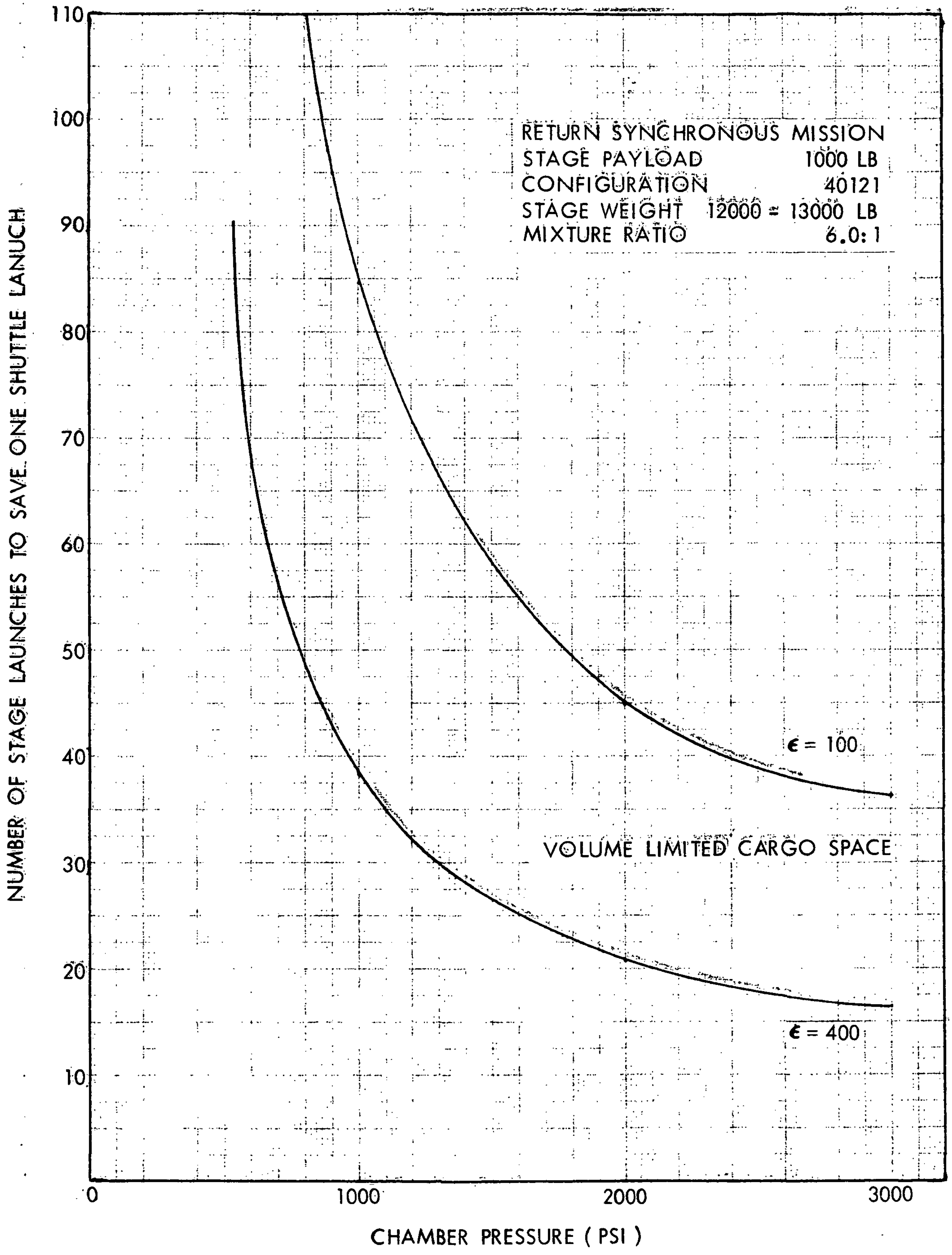

Figure 2-107. The Number of Stage Launches Required to Save One Shuttle Flight (12-13,000-LB Stage)

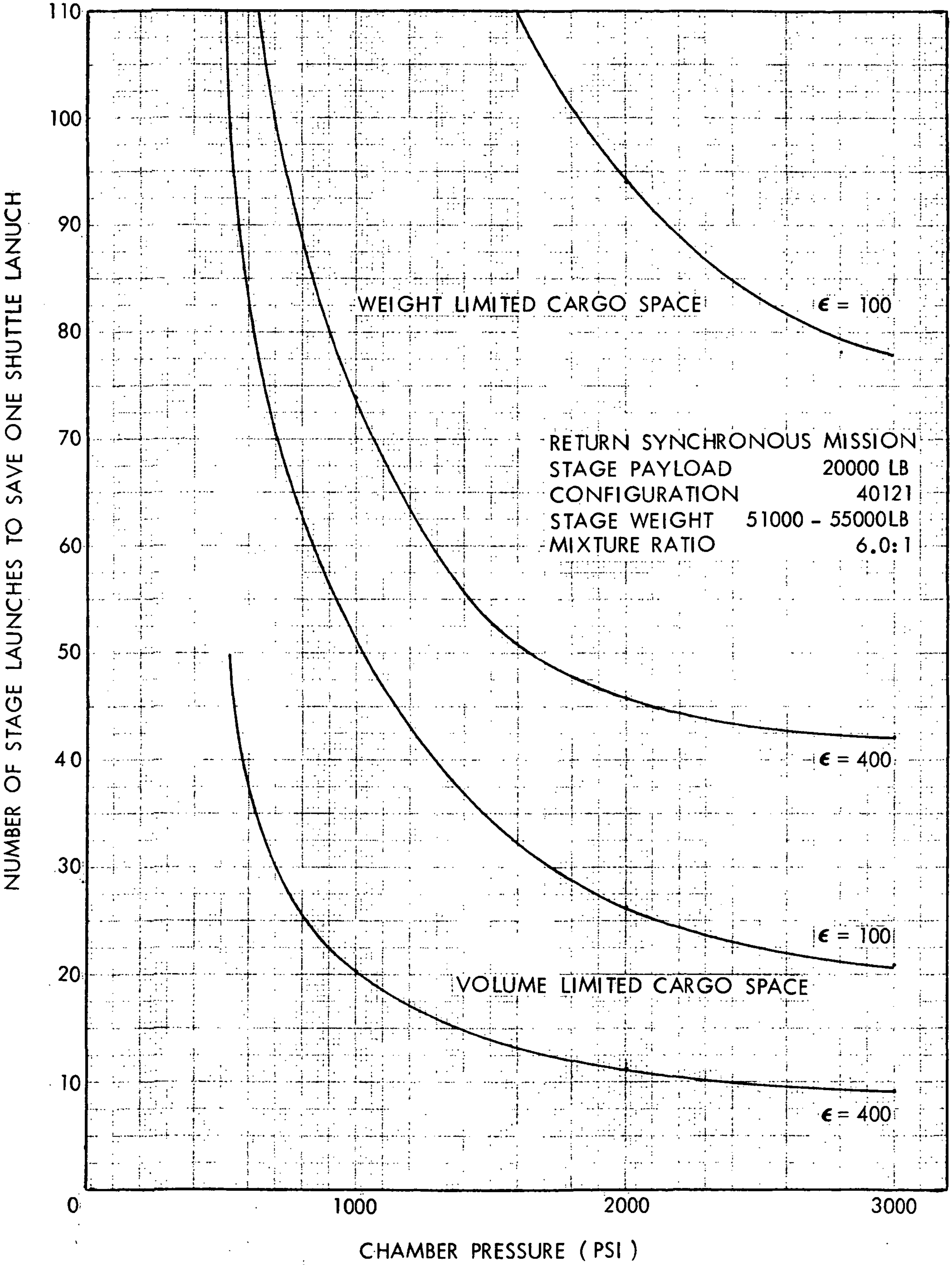

Figure 2-108. The Number of Stage Launches Required to Save One Shuttle Flight (51-55,000-LB Stage)

The number of shuttle plus stage launches required to break-even (technology investment costs = savings) are depicted in figures 2-109 through 2-111, for the two stage sizes, various technology investment costs and various expenditure rates for technology investment costs. As indicated in these figures, approximately 6 to 7 shuttle plus stage launches would be required before a cost of $5 million for higher chamber pressure technology would be made up by the savings realized through the use of smaller stages. The unrealistic volume-limited cargo space case, figure 2-109, indicates that net return will occur after only 2 to 3 flights.

In computing the technology break-even points, it was found that the stage investment and propellant cost savings (Δ Program - Δ RDT&E) constituted only a small part of the total savings which might be realized. That is, the stage RDT&E and shuttle launch savings made up almost the entire savings. Whether the stage RDT&E or shuttle savings were dominant, depends upon the level of technology investment and the chamber pressure. The savings in stage RDT&E are the overriding factor at the lower technology investment costs and the higher chamber pressures.

2.4.6 Conclusions Concerning Moderate Chamber Pressure Increases

The results of this analysis indicate that the cost of developing the technology required for moderate chamber pressure can probably be recovered through stage and program savings. However, the net total savings will be marginal. These data also show that the possibility for recovering the technology development cost increases with both stage size, engine nozzle expansion ratio, and shuttle launch costs. This last fact is the consequence of the savings in shuttle launch costs being an order of magnitude larger than the potential savings which might be realized in stage investment and operations cost (Δ Program - Δ RDT&E).

Although some benefit in the form of monetary savings will be derived from increased chamber pressure, the major justification probably will be dictated by other constraints; such as the need for additional performance from higher chamber pressures, to perform a certain design mission.

2.5 EFFECT OF REDUCTION OF ENGINE WEIGHT

While the analysis of moderate chamber pressures was being conducted, a study was undertaken to assess the implications that reductions in engine weight would have on stage and program costs. Three different synchronous mission stage sizes were considered for various reductions in engine weight ranging from 0 to 30 percent. The results obtained are discussed in the subsequent portions of this subsection.

2.5.1 Data and Assumptions

Throughout the study, certain constraints, guidelines and pertinent design data were used which are summarized in this section. Table 2-23 gives the design constraints used. The prime structure data used in computing the shell and thrust structure weights are shown in table 2-24. Table 2-25 summarizes the assumed tankage design data, including pertinent thermal and meteoroid protection data. The weights assumed for the astrionics systems and other miscellaneous subsystems are given in table 2-26. These

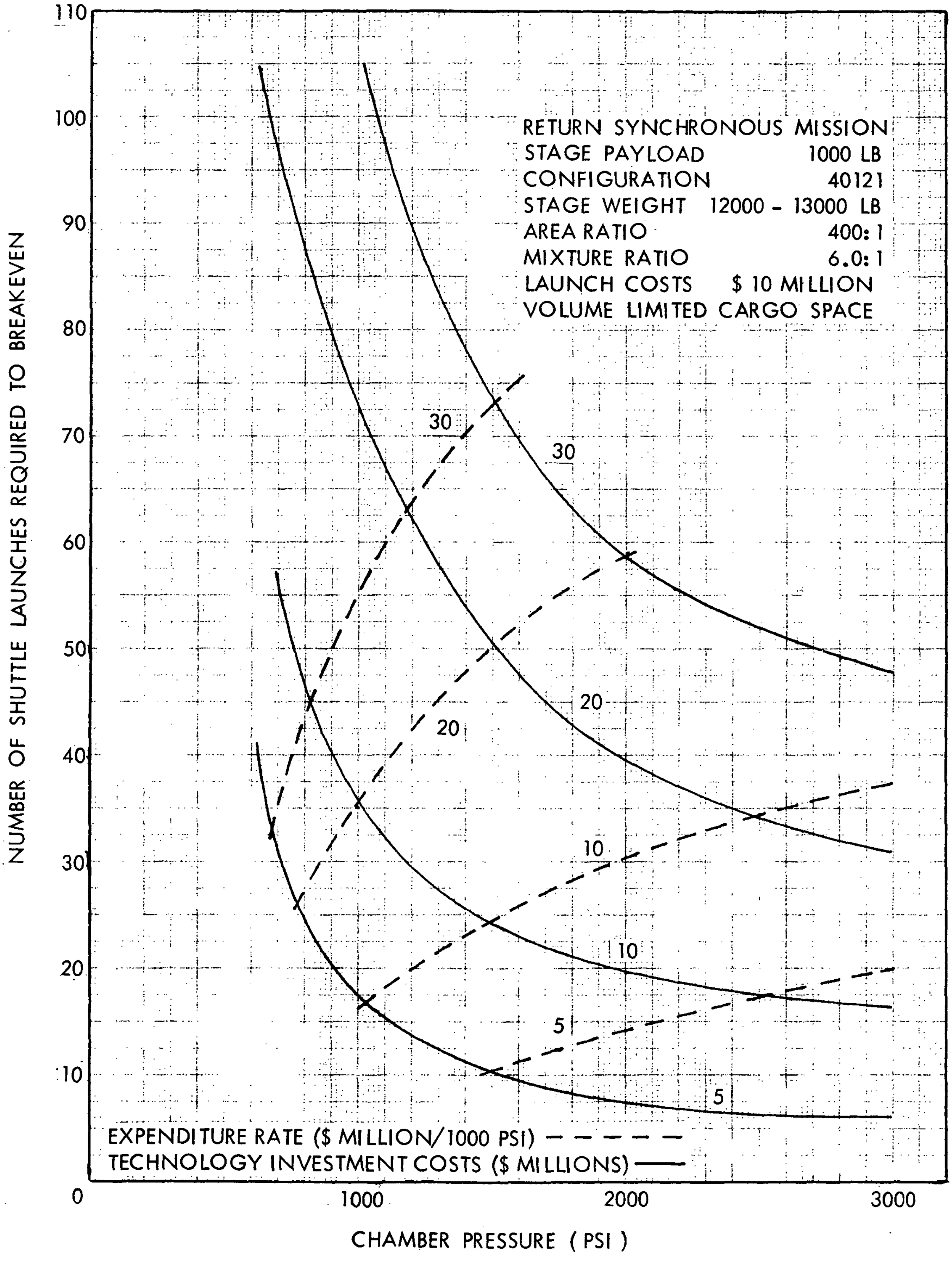

Figure 2-109. Break Even Flights for a 12-13,000 LB Stage (Volume-Limited Cargo Space)

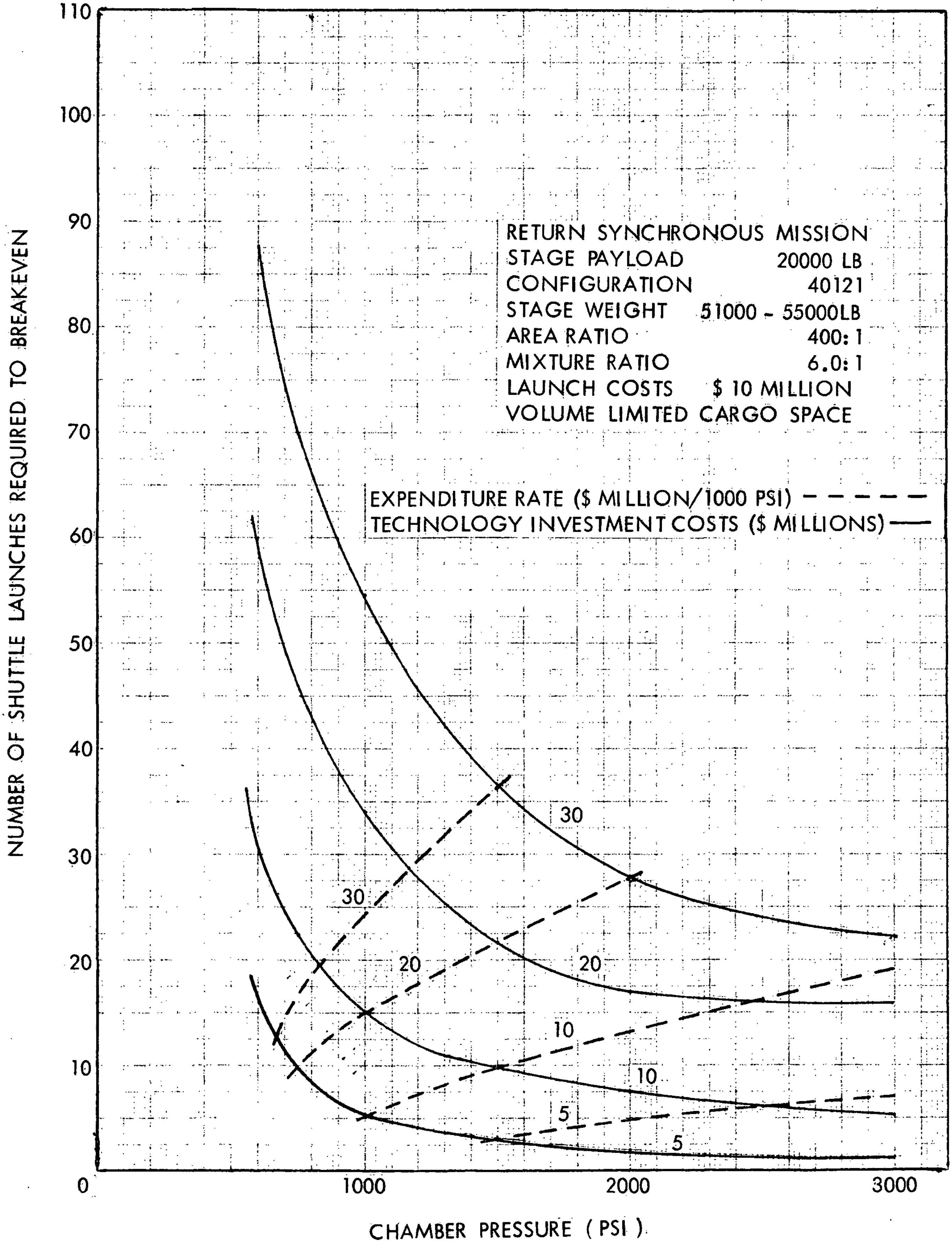

Figure 2-110. Break Even Flights for a 51-55,000 LB Stage (Volume-Limited Cargo Space)

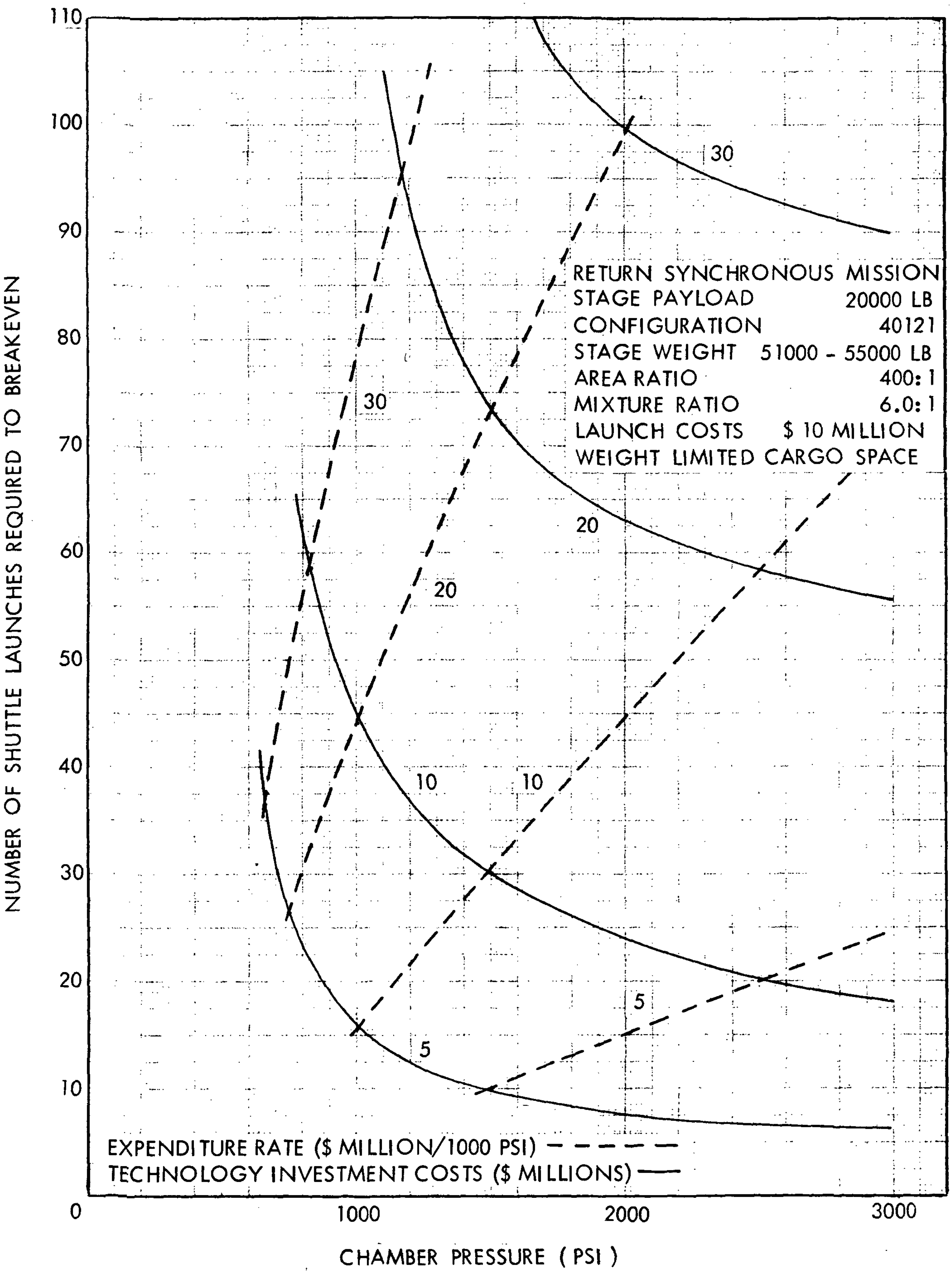

Figure 2-111. Break Even Flights for a 51-55,000 LB Stage (Weight-Limited Cargo Space)

Table 2-23. Summary of Stage Design Constraints

Constraint \ Mission	Single Stage Synchronous
Maximum Stage Diameter (In.)	174.0
Shell - Tank Spacing (In.)	6.0
Tank - Tank Spacing (In.)	6.0
Engine - Tank Spacing Factor (Chamber)	4.0
Engine - Tank Spacing Factor (Exit)	0.8
Engine - Booster Spacing (In.)	0.0
Engine Gimbal Angle (Degrees)	3.0
Thrust - To - Weight Ratio	0.25
Axial Acceleration (G's)	1.00
Lateral Acceleration (G's)	0.05
Payload Density (Lb/Ft^3)	25.0
Inert Weight Contingency Factor (%)	7.5

Table 2-24. Summary of Structural Design Data

Data \ Structure	Shell	Thrust Cone
Material	Aluminum	Aluminum
Density (lb/ft^3)	183.0	183.0
Material Strength (psi)		
Tension	67,000	67,000
Compression	46,000	46,000
Modulus of Elasticity (psi)	10^7	10^7
Safety Factors		
Tension	1.25	1.25
Compression	1.00	1.00
Monocoque-to-Complex Structure Weight Ratio	*	*
Spider Beam Multiplication Factor	N/A	N/A

* A function of diameter and limit load; see appendix C.

Table 2-25. Summary of Tankage Design Data

Data \ Mission	Synchronous
Tankage	
Material	Aluminum
Density (lb/ft^3)	183.0
Allowable Stress (psi)	60,000
Factor of Safety	1.10
Minimum Skin Gauge (In.)	0.025
Land Factors (Bulkheads)	0.10
Land Factors (Cylindrical Section)	0.05
Thermal Protection	
Initial Fuel/Oxidizer Temperature (°R)	36.0/162.6
Initial Fuel/Oxidizer Pressure (psi)	15.0/15.0
External Insulation Temperature (°R)	450.0/470.0
Insulation Density (lb/ft^3)	4.5
Insulation Thermal Conductivity (Btu/Hr-Ft-°R)	*
Meteoroid Protection	
Probability of no Punctures	0.995
Nominal Mission Altitude (n.m.)	200
Shield Material	Aluminum
Material Density (lb/ft^3)	183.0
Material Yield Stress (psi)	70,000
Minimum Skin Gauges (In.)	0.015
Miscellaneous	
Minimum Fuel/Oxidizer Ullage Volume (%)	5.0/5.0
Residual Fuel/Oxidizer Fraction (%)	2.0/2.0
Feedline Flow Velocity (fps)	20.0
Tank Support Factor	ξ

* A function of temperature and thickness; see appendix C.
ξ Dependent upon configuration; see appendix C.

Table 2-26. Miscellaneous Subsystem Weights (MSFC Tug)

Electric Distribution	200 lb
Electric Power Power Systems Fuel Cell Reactants	300
Communication/Instrumentation Communications Data Management Instrumentation	295
Guidance, Navigation, and Control Guidance, etc. Rendezvous and Docking Radar	210
Hydraulic/Pneumatic Purge Umbilical Tug/Orbiter Service	145
Propellant Utilization	35
Miscellaneous Destruct System Docking Adapter Subsystem Mounts Orbiter Interface Payload Interface Purge	1515
Total	2700 lb

weights reflect the miscellaneous subsystem philosophy being considered in an in-house MSFC Space Tug study(1).

The stage geometry selected as the baseline for this analysis was the 40121 configuration, which has a large single hydrogen tank with ellipsoidal domes; and four small oxygen tanks with hemispherical bulkheads, suspended below the thrust cone. This stage geometry was selected instead of the tandem tank configuration because the shorter stage would be more advantageous for use in the shuttle's cargo bay. A typical stage having this configuration is illustrated in figure 2-112.

The parametric oxygen-hydrogen engine system performance, weight and geometry data used in this study, were obtained from Rocketdyne for use in the "LOX/Hydrogen Engine Technology for Advanced Missions" study, contract NAS7-790. These data covered engines utilizing topping, expander and gas generator cycles. The data for the topping and expander cycles included thrust levels from 15,000 to 120,000 pounds, area ratios from 100 to 400, and mixture ratios of 5.0, 6.0 and 7.0. Data were supplied for chamber pressures of 500 to 1000 psi, and 1000 to 3000 psi, for the expander and topping cycles, respectively. The gas generator cycle data covered lower thrust engines (5,000 - 15,000 lb) and chamber pressures of 800 and 1000 psi. These parametric engine data were published in appendix B of the final report for the study cited above.

The costs generated during this analysis were based on cost estimating relationships which are predicated on historical cost data and pertinent vehicle parameters.(4) In general, the cost estimating relationships of any cost element contain coefficients which indicate the technology level and complexity of that individual element. The technology level assumed for the main systems on the stage is presented in table 2-27. Table 2-28 lists the percent learning curves used to compute the investment costs.

The program cost data developed during this analysis include only the RDT&E, investment and upper stage propellant costs. The program costs presented do not contain the other cost elements normally included in the operations costs, because they are mainly launch vehicle and mission model oriented, and for identical missions and similarly sized upper stages the operational costs are relatively insensitive to variations in upper stage weight. Hence, any program cost sensitivities which are determined from these program cost data will be of sufficient accuracy.

2.5.2 Mission Profile

The mission profile selected for this analysis was the return of a variously sized payload from synchronous orbit. This mission profile, illustrated in figure 2-113, is similar to one being considered for a MSFC Space Tug. This two-burn mission would require the liquid oxygen/liquid hydrogen stage to be launched and placed in synchronous orbit by another stage or by itself with the use of external drop tanks. That is, the stage has a full (internal) propellant load at the beginning of the synchronous orbit coast. The first burn utilizing the "on-board" propellants is the retro maneuver associated with the return Hohmann transfer and plane change at synchronous orbit. The second and final burn circularizes the stage into a low earth orbit.

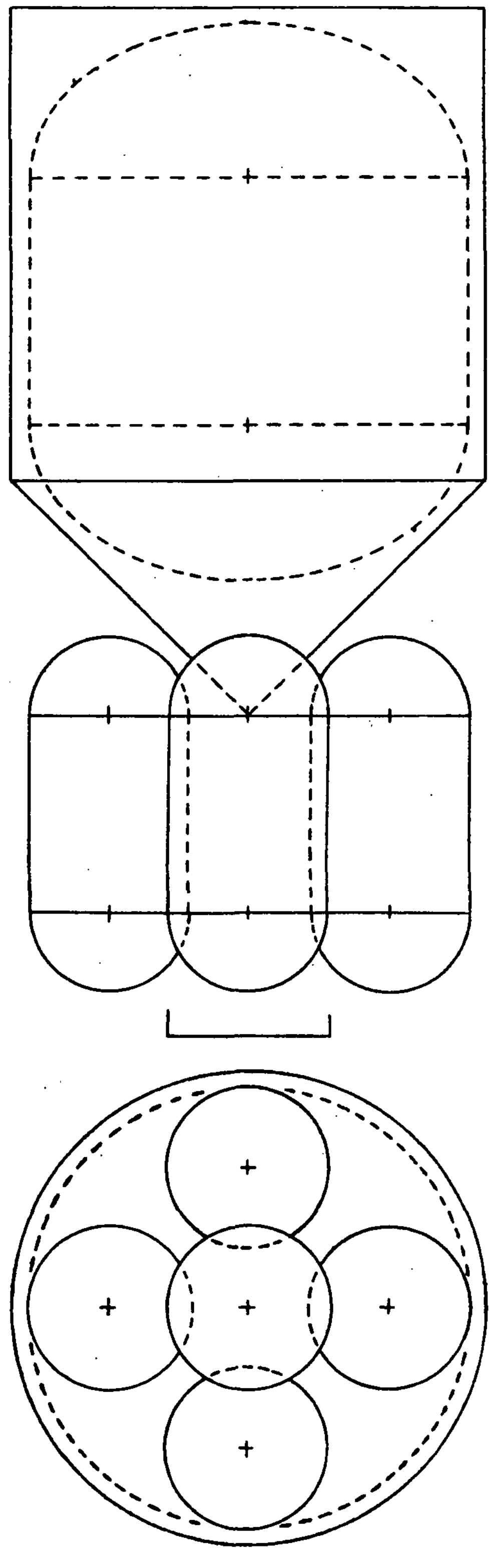

Figure 2-112. Multiple Oxidizer Tank Stage Configuration (40121)

Table 2-27. Technology Level of Systems

Area	Technology Level and Technology*
Structures	
Shell	SOA - Aluminum Sheet Stringer
Thrust Structure	SOA - Aluminum Sheet Stringer
Tankage	SOA - Aluminum Monocoque
Meteoroid Shield	SOA - Aluminum Monocoque
Tank Supports	ADV - Composite
Propellant Feedlines	SOA - Aluminum
Propulsion	
Main Engines	New, Advance, Reuseable LH_2/LOX
Reaction Control Thrusters	SOA - Monopropellant
Miscellaneous Subsystems	Adaptation of existing hardware to a new unmanned, reuseable upper stage
Electrical Power and Distribution	
Electrical	
Communication	
Instrumentation	
Guidance, Navigation and Control	
Hydraulic/Pneumatic	
Propellant Utilization	
Destruct	

* SOA - State-of-the-art Technology
ADV - Advance Technology

Table 2-28. Investment Learning Curves

System	Learning Curve
Structures	90%
Propulsion	95%
Miscellaneous Subsystems	90%
Assembly and Checkout	90%

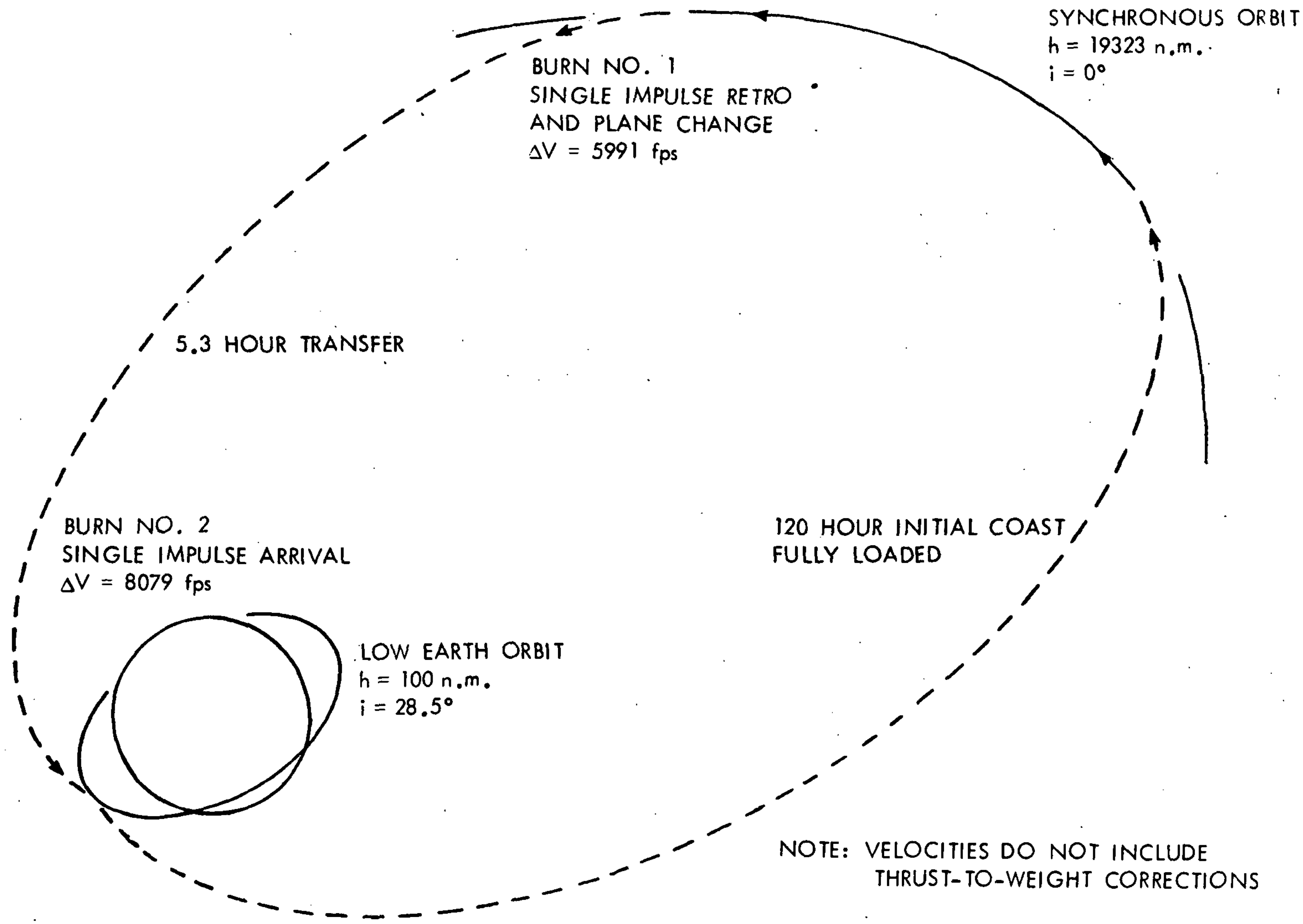

Figure 2-113. Mission Profile for Return Synchronous Mission

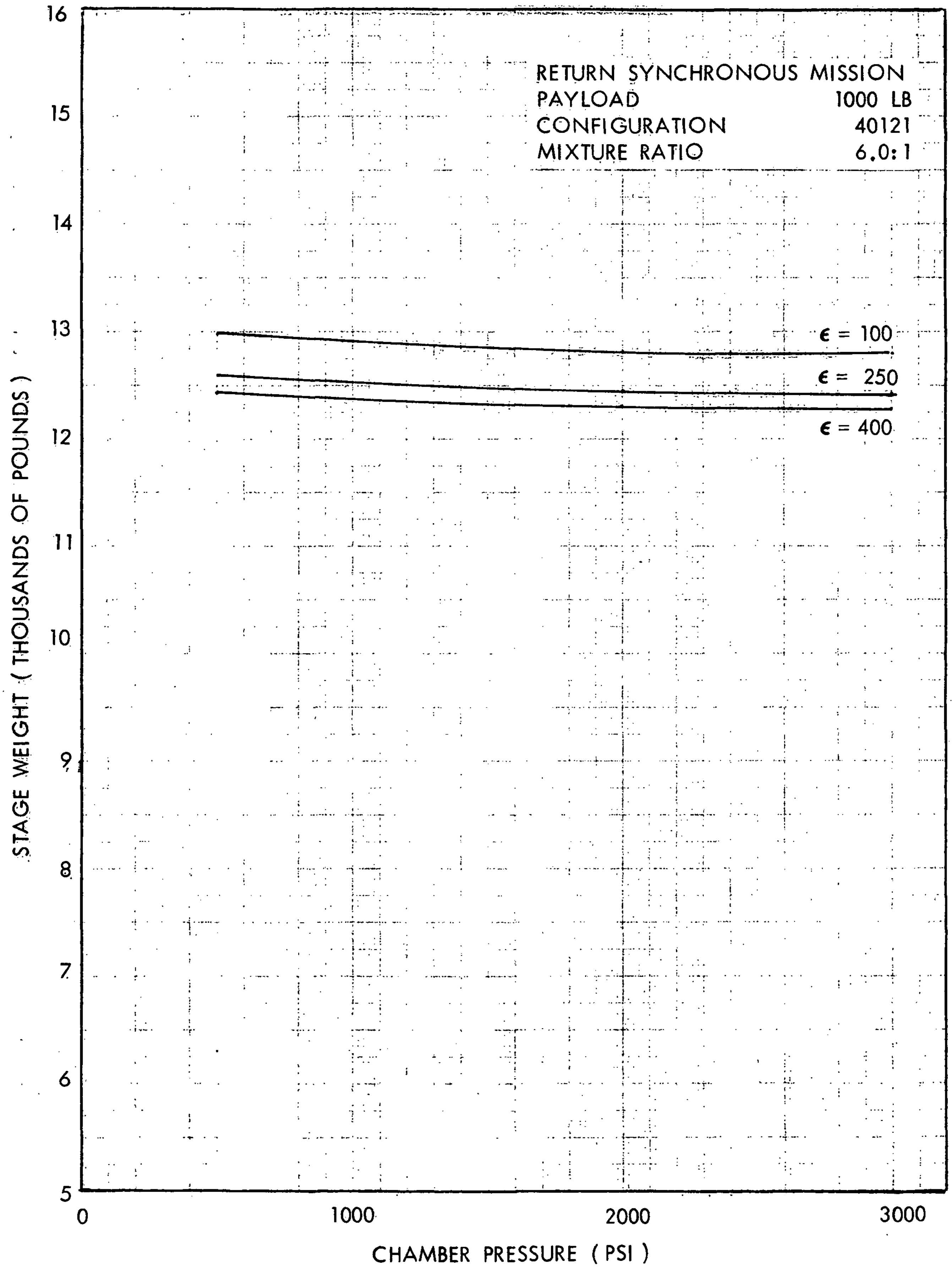

Figure 2-114. Stage Weights for a 1000-LB Payload

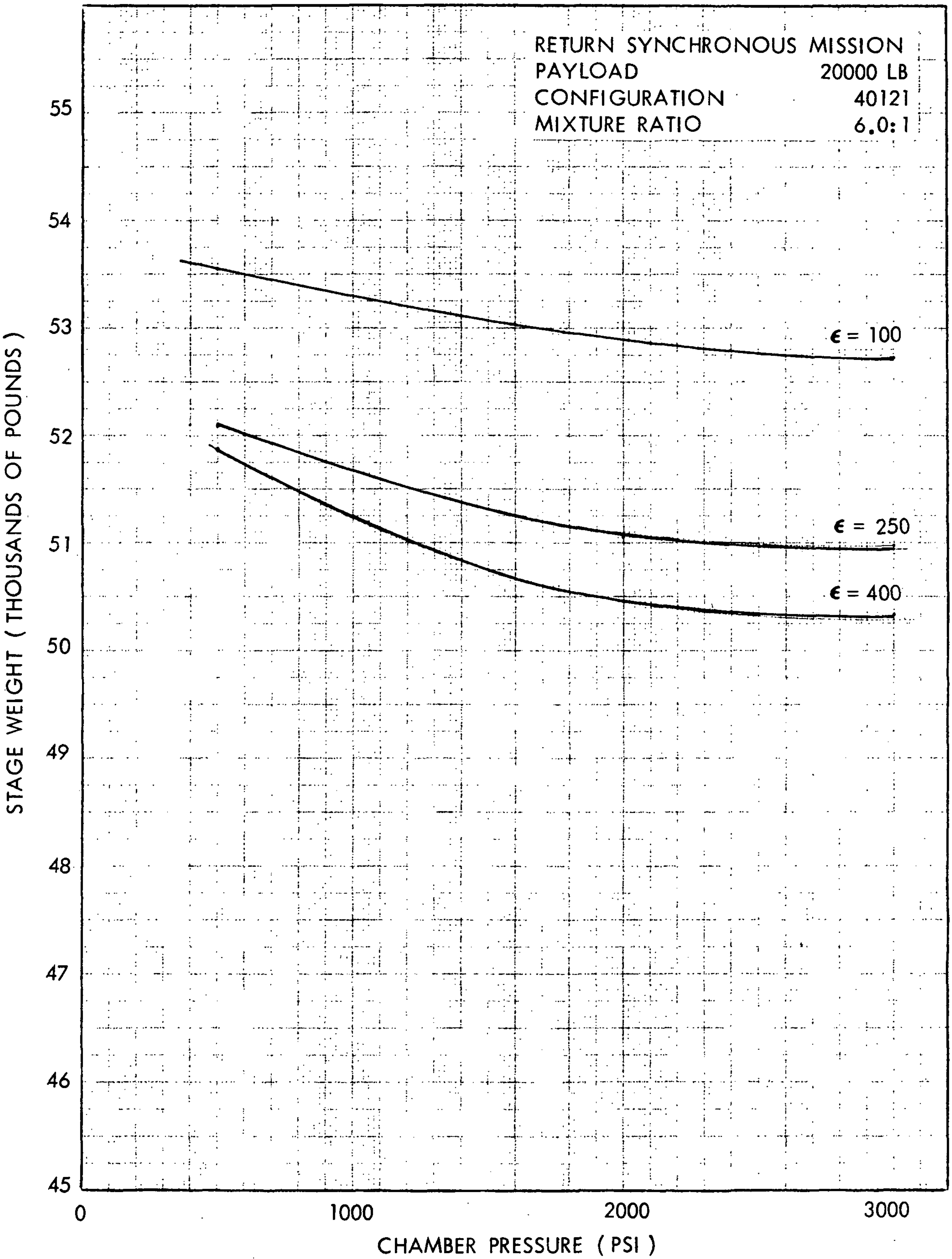

Figure 2-115. Stage Weights for a 20,000-LB Payload

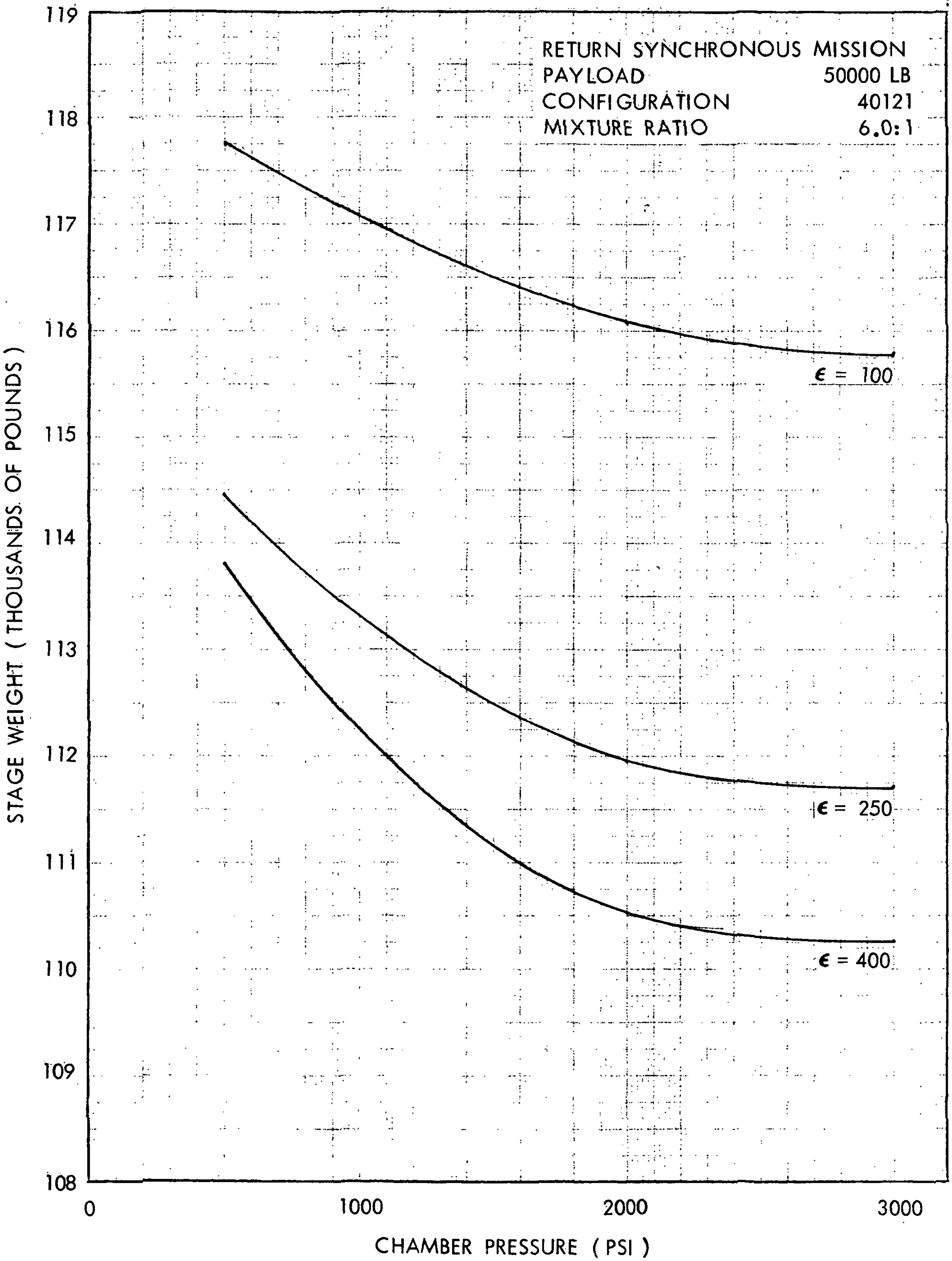

Figure 2-116. Stage Weights for a 50,000-LB Payload

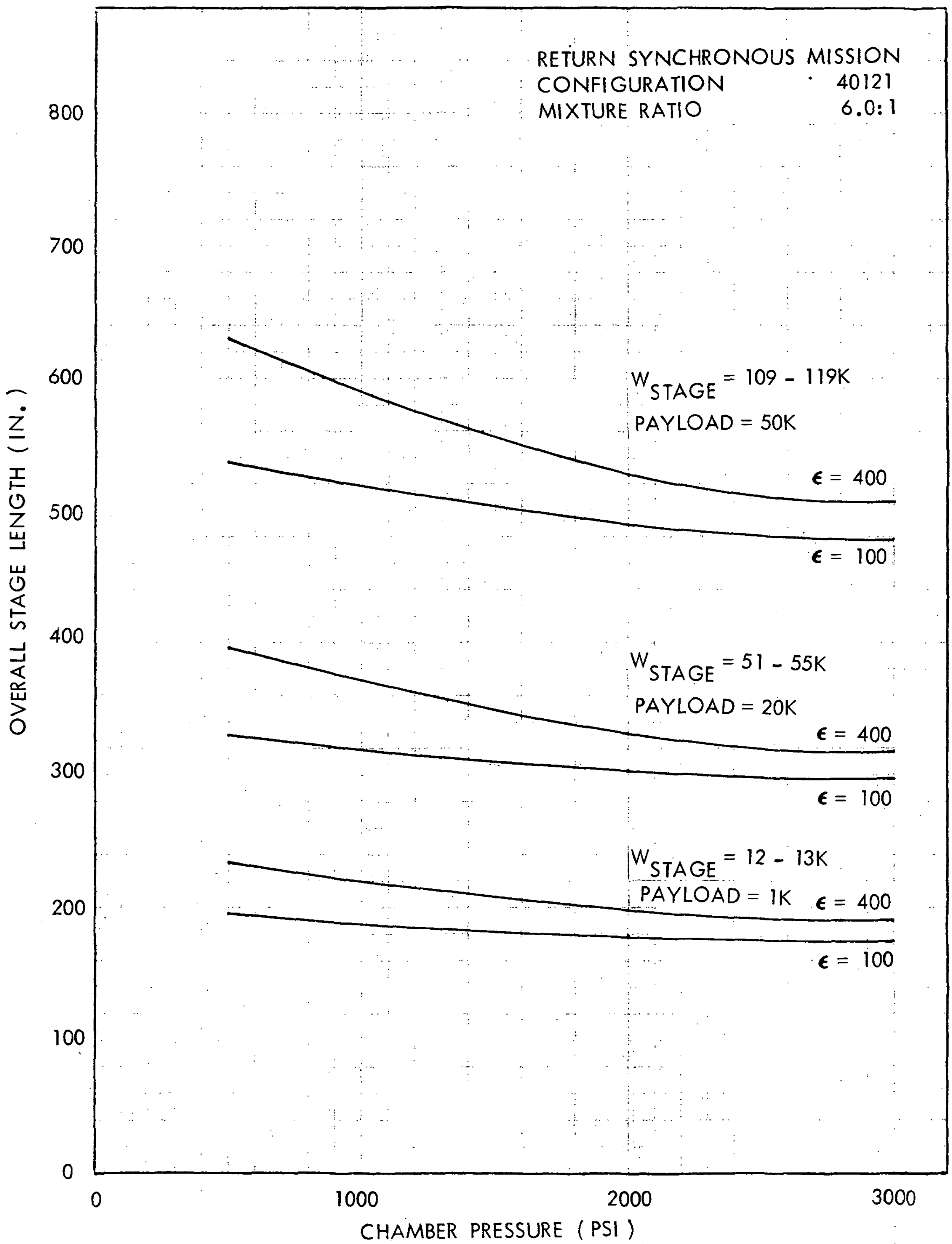

Figure 2-117. Over-all Stage Length

The velocities used for this mission assume a Hohmann transfer between an equatorial (0-degree inclination) synchronous orbit, and a 28½-degree inclination, circular 100 nautical mile orbit. The velocities were not corrected to account for the effects of the stage's initial thrust-to-weight ratio (T/W = 0.25) and specific impulse, nor were the effects of orbital regression on the velocity requirements considered.

2.5.3 Stage Size and Cost

Variously sized stages were analyzed with engines having chamber pressures ranging from 500 to 3000 psi and nozzle expansion ratios of 100:1, 250:1 and 400:1. The results of the sizing analyses are presented in figures 2-114 through 2-116. The stage weights which correspond to various engine area ratios are depicted as a function of chamber pressure for stages capable of carrying payloads of 1000, 20,000 and 50,000 pounds. These data are based on nominal engine weights; that is, 100 percent the normal engine weight.

The overall length of the states investigated is presented as a function of chamber pressure (figure 2-117).

The RDT&E, TFU and program costs were determined for each of the stages sized. The costs were computed with a computer routine, developed specifically for upper stages, which determines costs from cost estimating relationships.[4] The resulting RDT&E, TFU and program costs are depicted for various stage sizes in figures 2-118, 2-119, and 2-120, respectively.

The 20-stage program costs shown in figure 2-120 include only the RDT&E, investment, and upper stage propellant costs. Other costs which are normally included in operations have not been included (see subsection 2.5.1).

2.5.4 Variations in Stage Size and Cost

The differences in stage size and cost for each of the three stages (1,000; 20,000 and 50,000-pound payload capabilities) were determined for various percent reductions in engine weight. The results, depicted in figures 2-121 through 2-135, show the reductions in stage size and cost which accompany decreases in engine weight.

The reductions in stage weight and length, which result from reduction in engine weight, are presented in figures 2-121 and 2-222, respectively, for a 12,000 to 13,000-pound stage having a 1000-pound payload capability.

The RDT&E, TFU and program cost savings estimated for these 12-13,000 pound presented as a function of percent reduction in engine weight, in figures 2-123, 2-124 and 2-125, respectively. These data are presented for several different combinations of engine chamber pressures and area ratios. The costs associated with developing the lighter engines are not reflected in these data.

Similar data are presented for two differently sized stages in figures 2-126 through 2-135. Data for a 51-55,000-pound stage (20,000-pound payload capability) are presented in figures 2-126 through 2-130; and for a stage having a payload capability of 50,000-pounds (109-119,000-pound stages) in figures 2-131 through 2-135.

The data indicates that regardless of stage size, only extremely small savings in stage weight, length and cost can be obtained through the reduction of engine weight.

2.5.5 Conclusions Concerning Reductions in Engine Weight

Although no analyses were conducted to determine the technology break-even points, the conclusion that reduction of engine weight would not be attractive from the stand point of cost savings is quite obvious. The reasons were twofold. First, the stage RDT&E cost savings that are realized through engine weight reductions are minimal at best. And second, the smaller stage size associated with the lighter engines will not yield a large savings in shuttle transportation costs. This last fact is obvious when the 1000-pound and 3-inch variations (maximum) in stage size obtained through engine weight reduction are compared to the 2,200+-pound and 120-inch differences found during the analysis of moderate chamber pressure increases (section 2.4). In addition, it is doubtful whether any overriding considerations (i.e., the requirement for mass ratio improvement by weight reduction in order to perform a necessary mission) can be found where the return on technology investment would not be higher if the funds were invested in other areas to correct the problem.

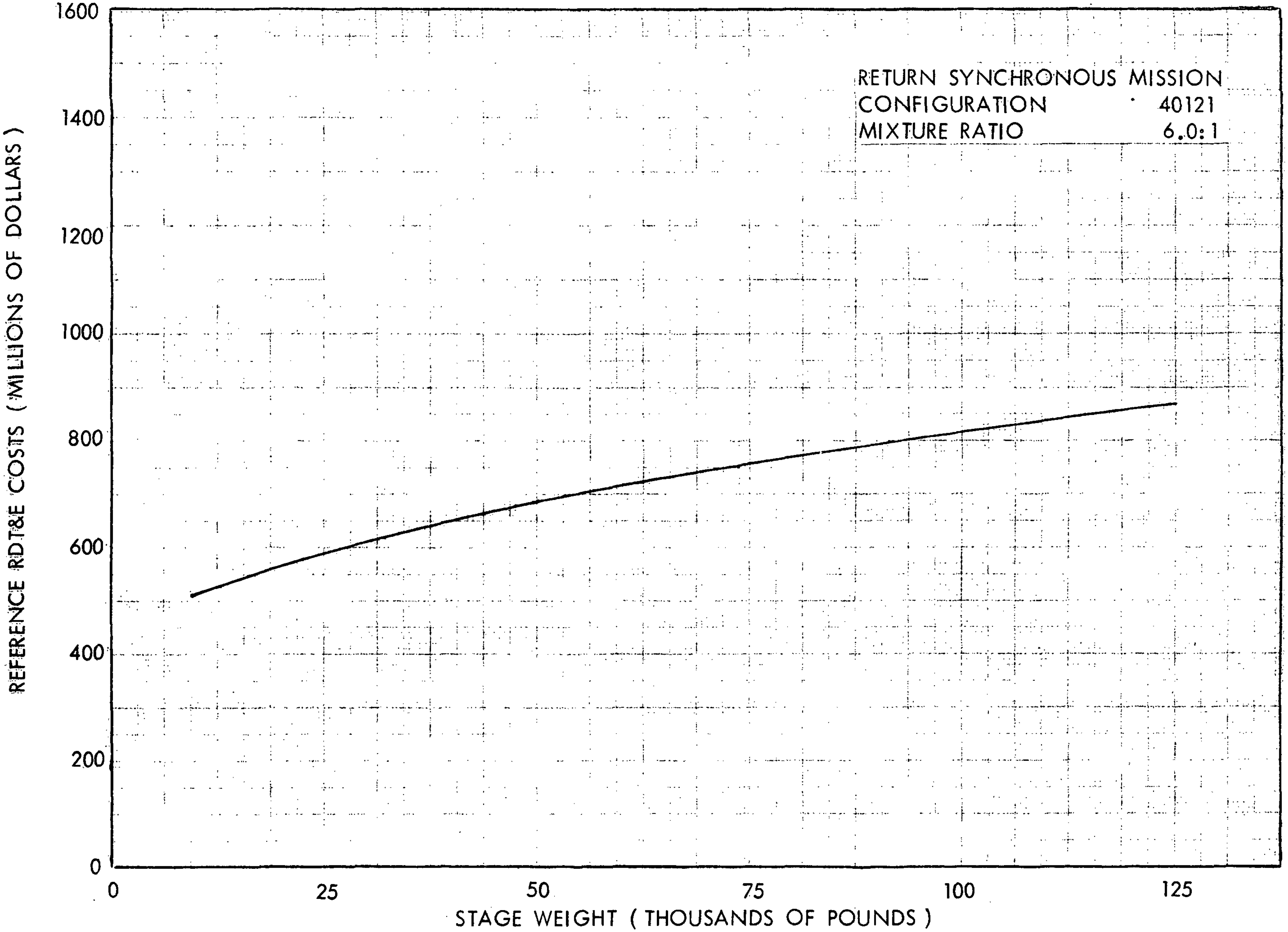

Figure 2-118. Reference RDT&E Costs

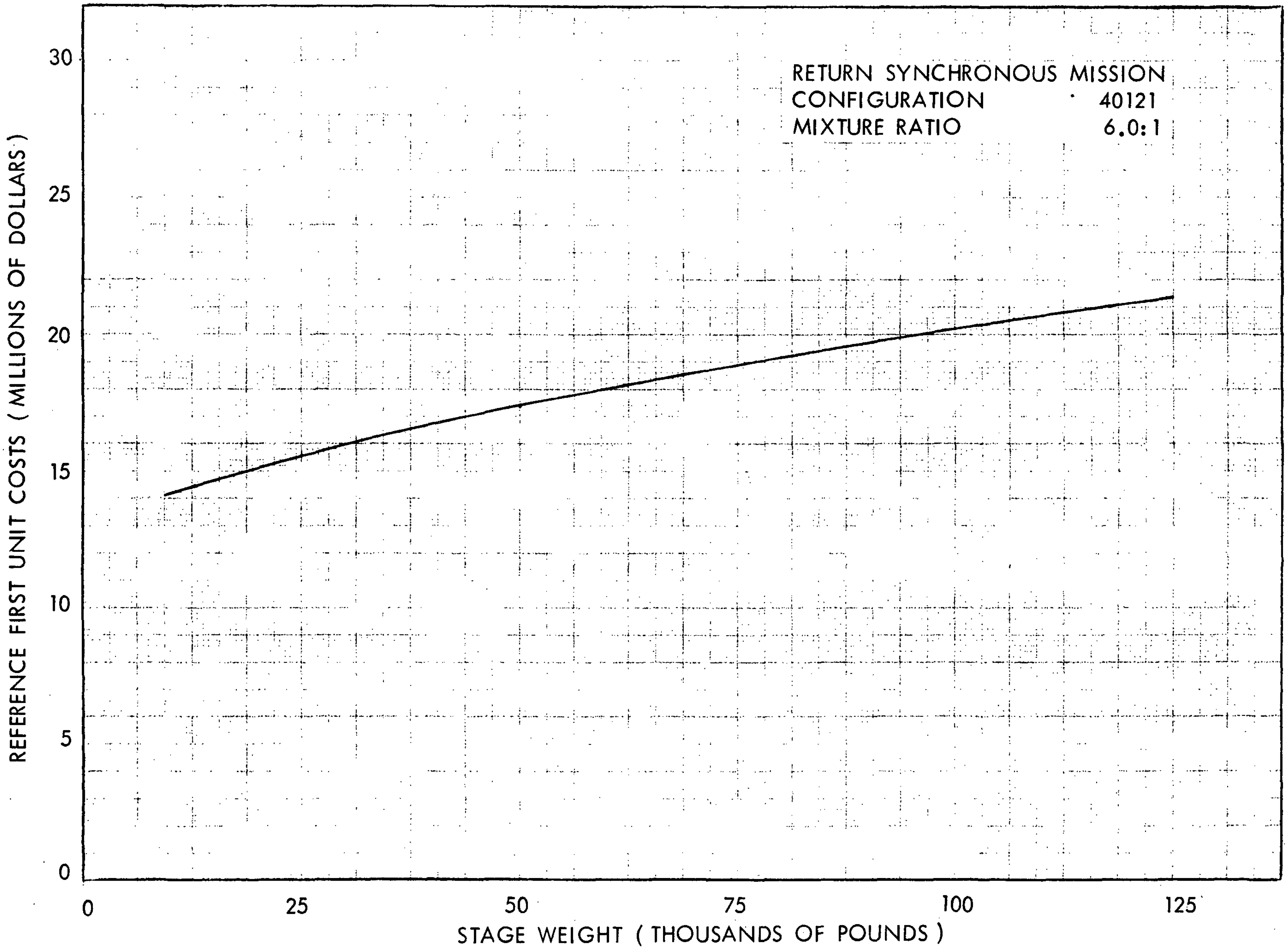

Figure 2-119. Reference TFU Costs

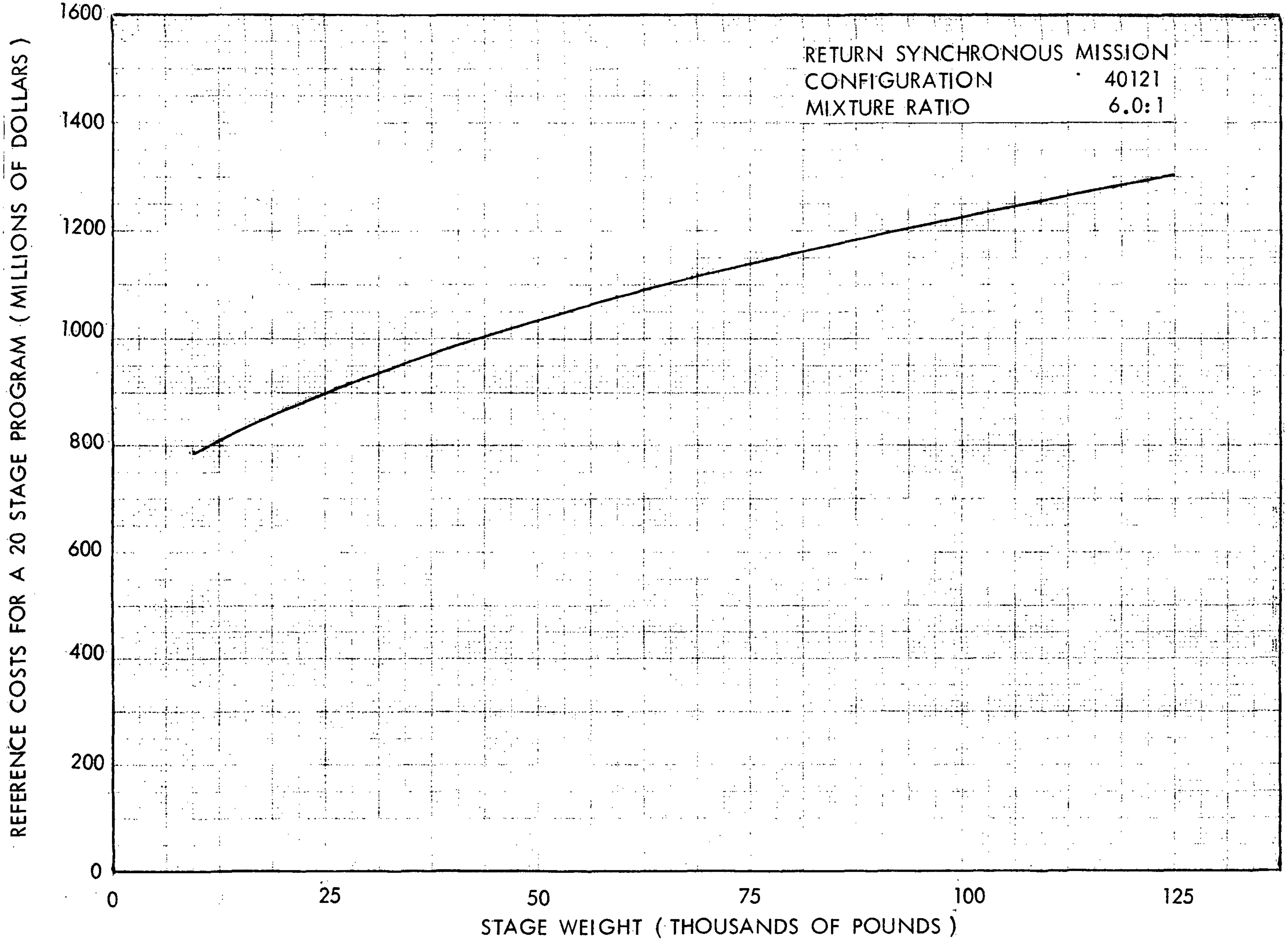

Figure 2-120. Reference Costs for a 20 Stage Program

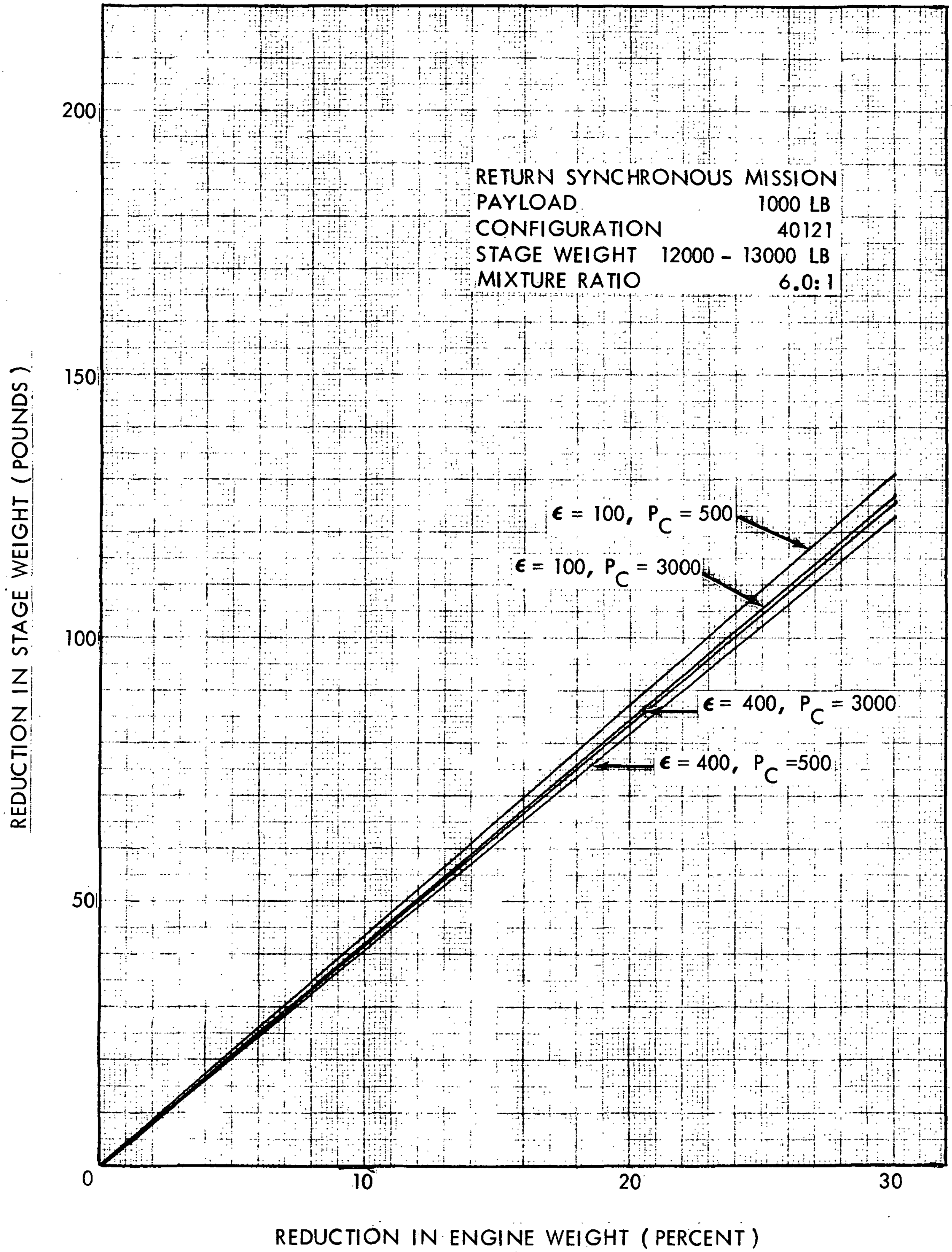

Figure 2-121. Reduction in Stage Weight (12-13,000-LB Stage)

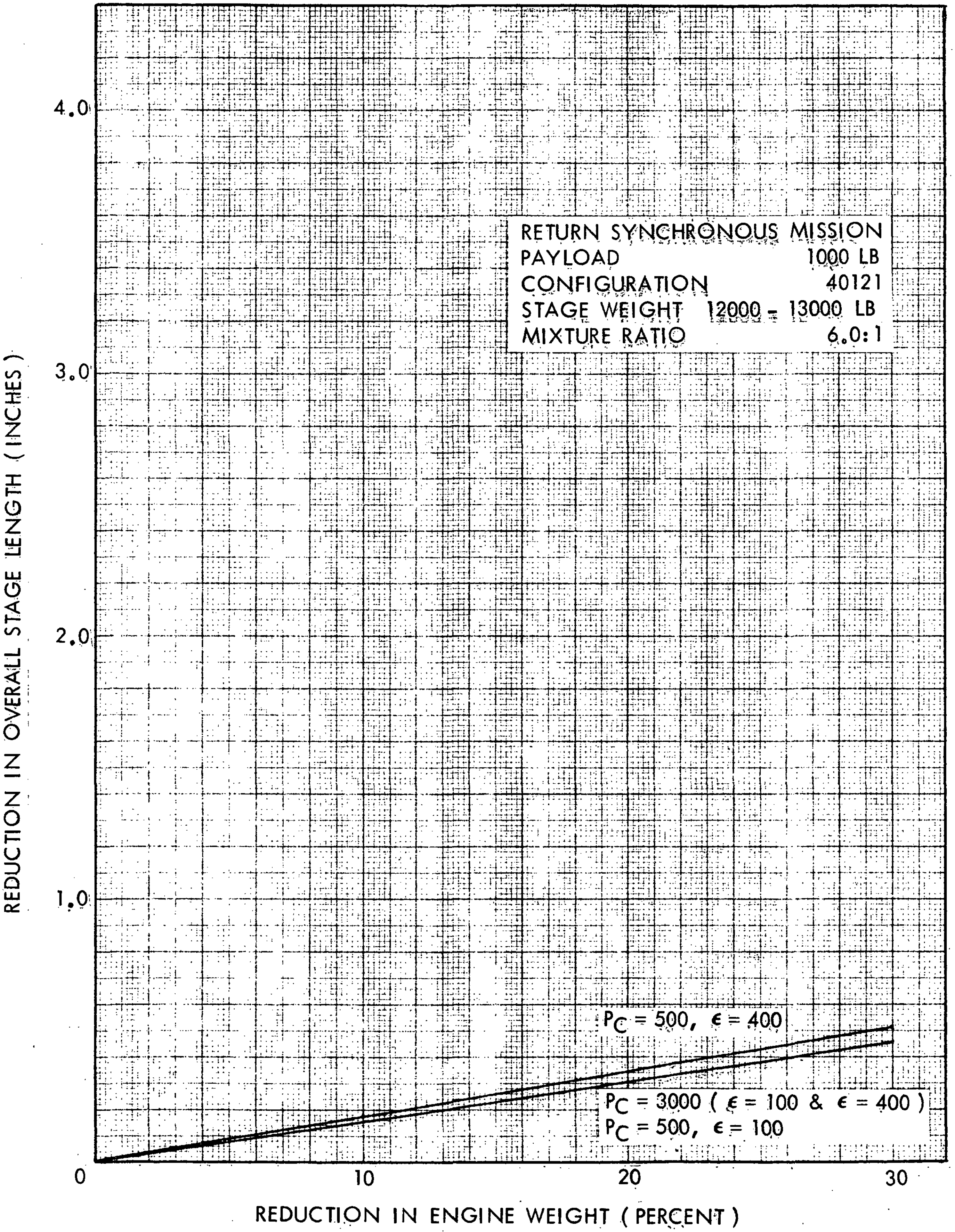

Figure 2-122. Reduction in Stage Length (12-13,000-LB Stage)

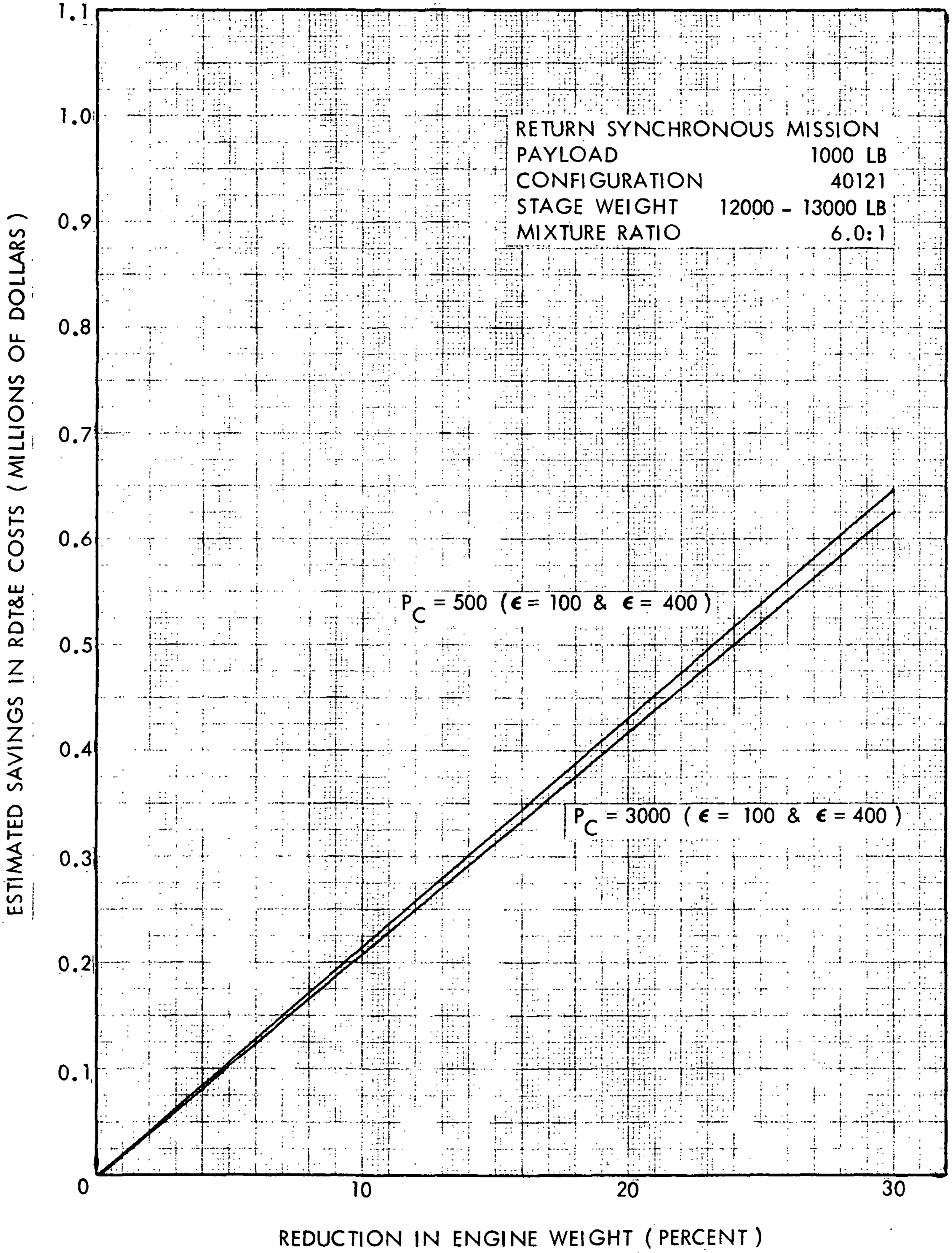

Figure 2-123. Estimated Savings in RDT&E Costs (12-13,000-LB Stage)

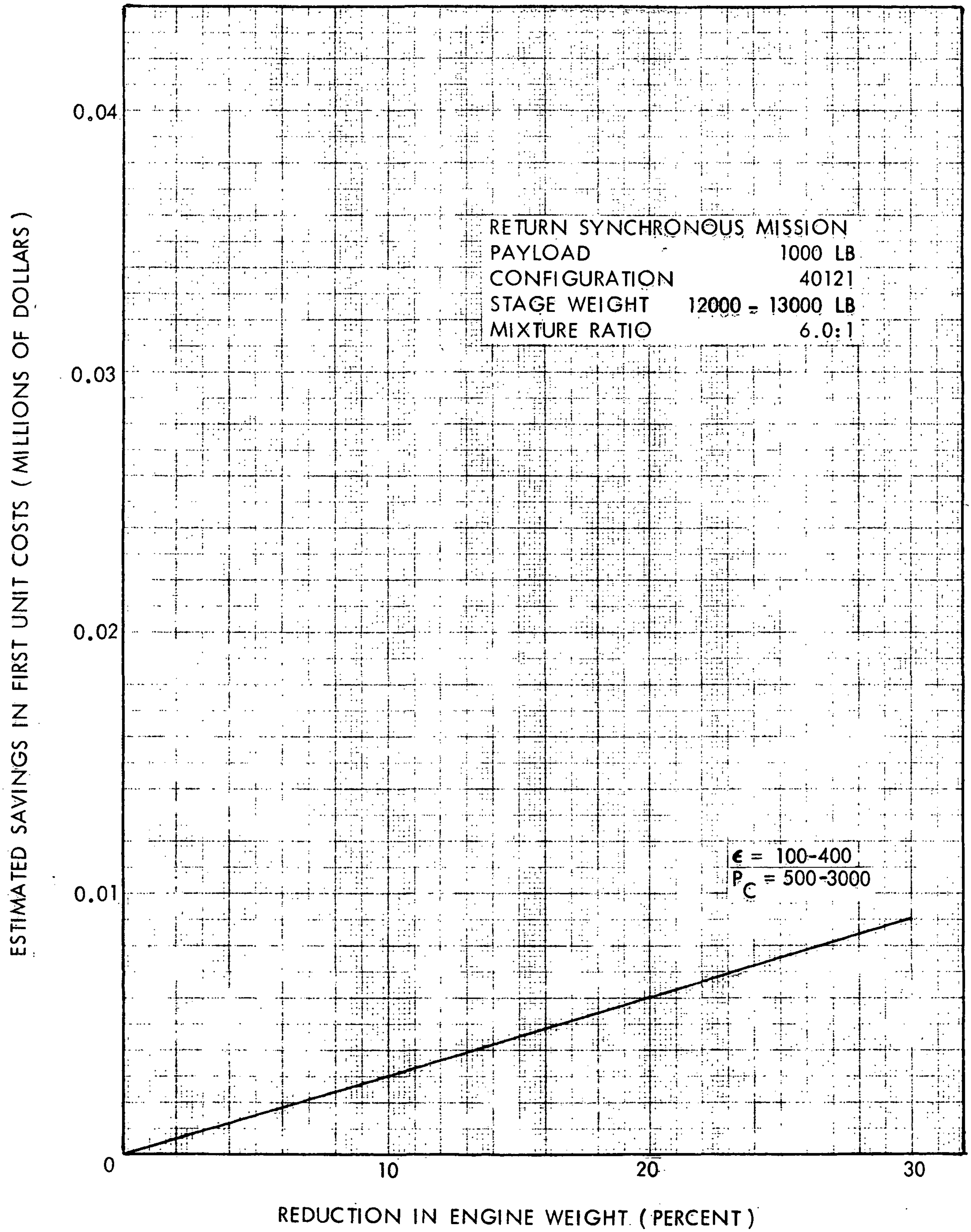

Figure 2-124. Estimated Savings in TFU Costs (12-13,000-LB Stage)

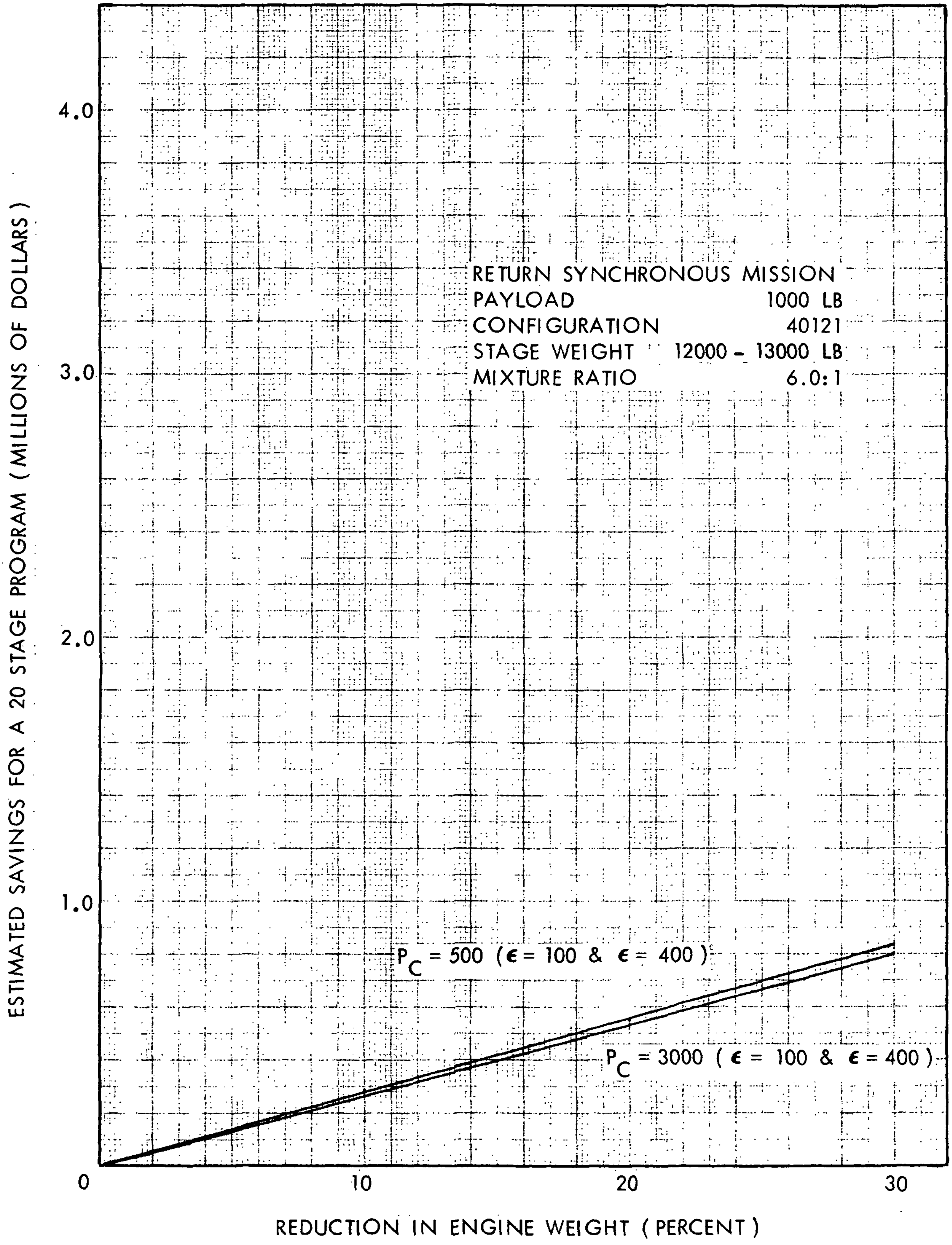

Figure 2-125. Estimated Savings for a 20-Stage Program (12-13,000-LB Stage)

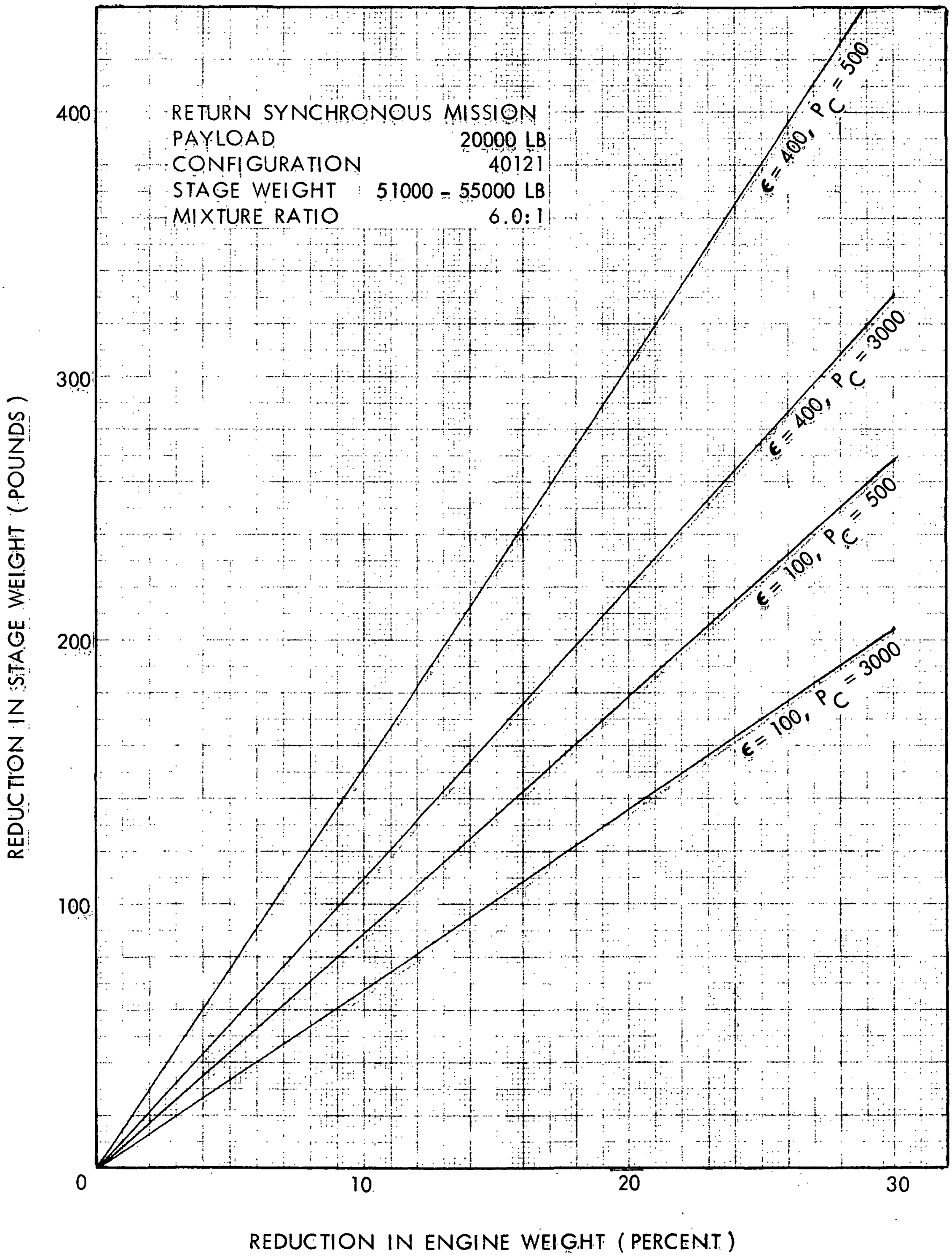

Figure 2-126. Reduction in Stage Weight (51-55,000-LB Stage)

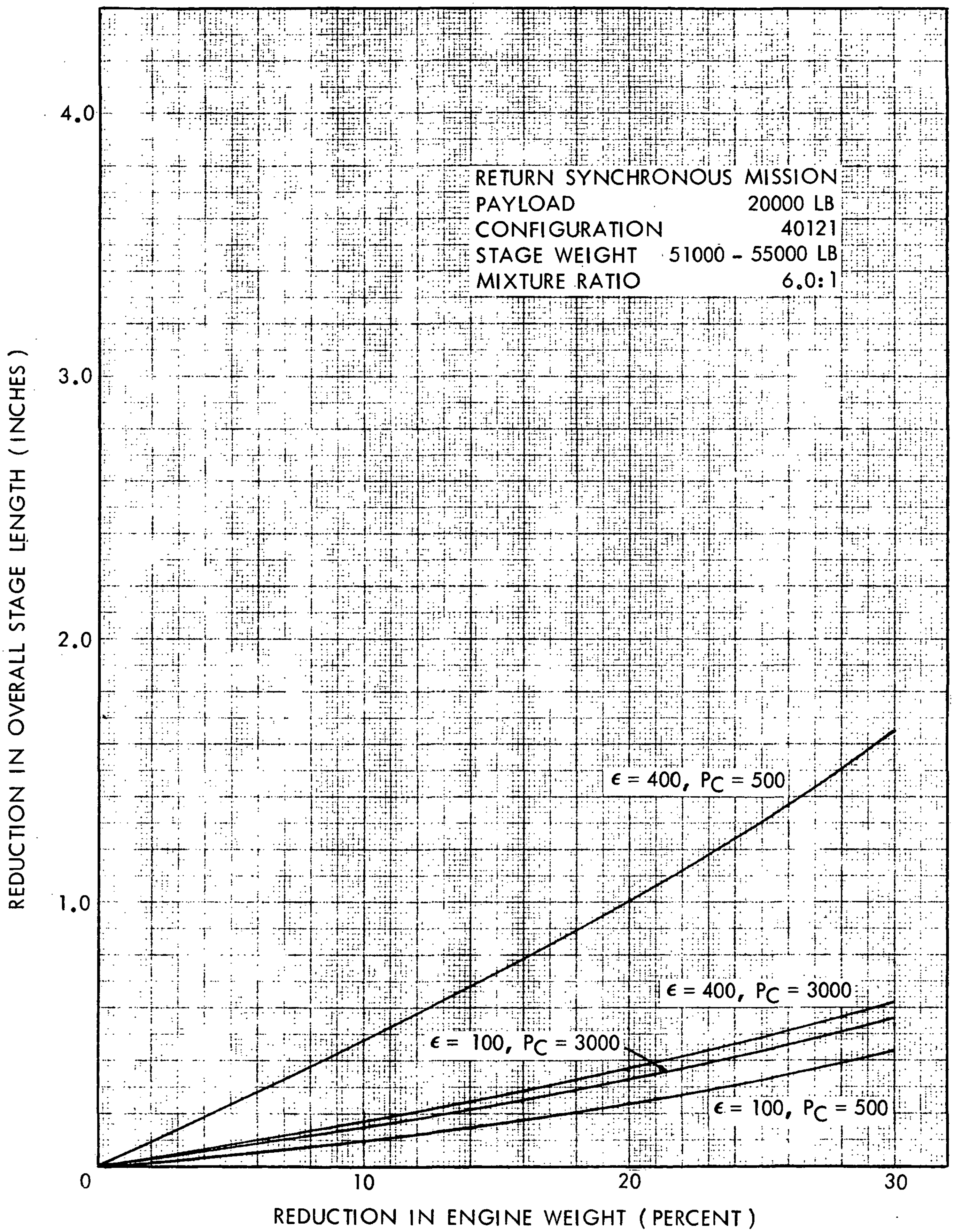

Figure 2-127. Reduction in Stage Length (51-55,000-LB Stage)

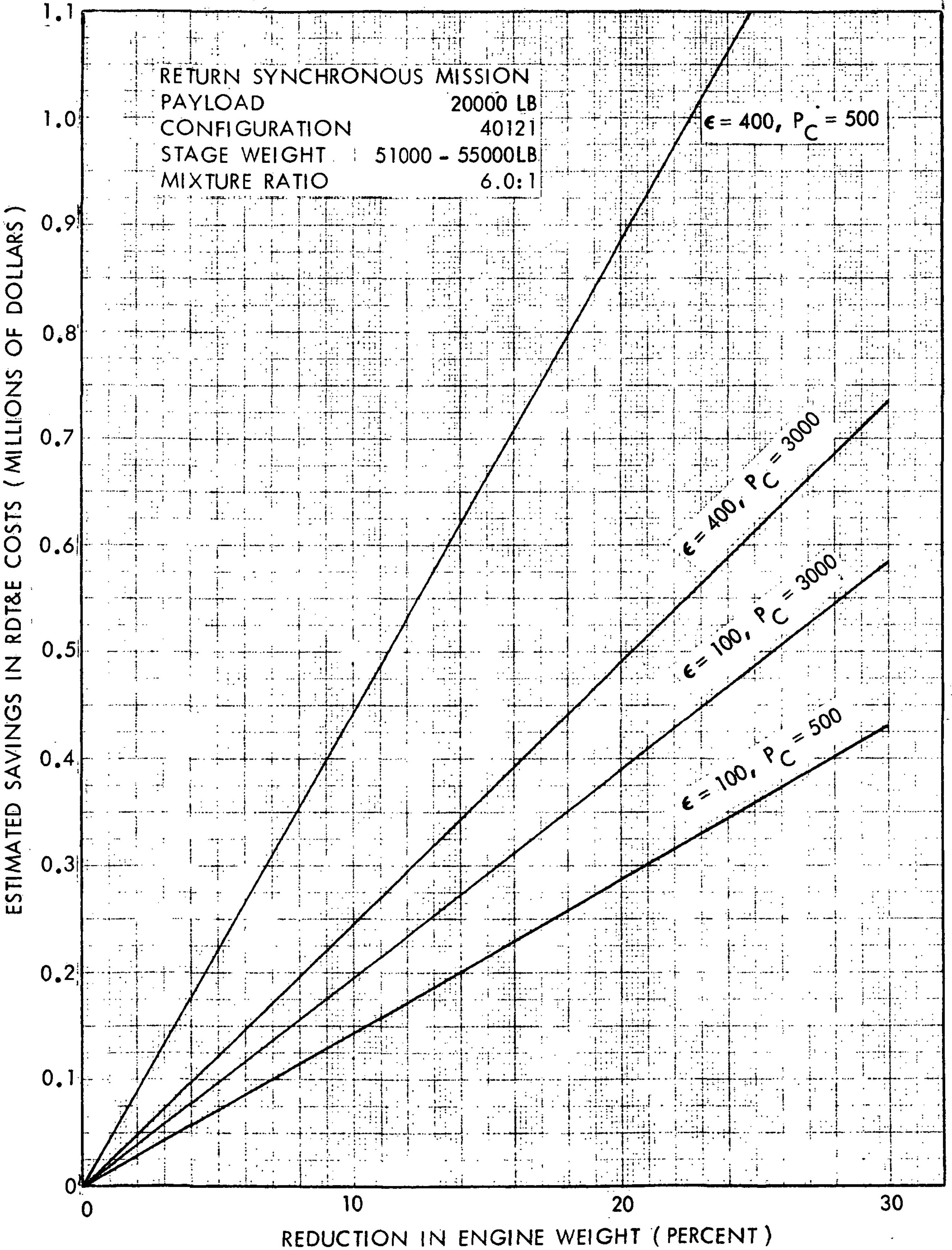

Figure 2-128. Estimated Savings in RDT&E Costs (51-55K Stage)

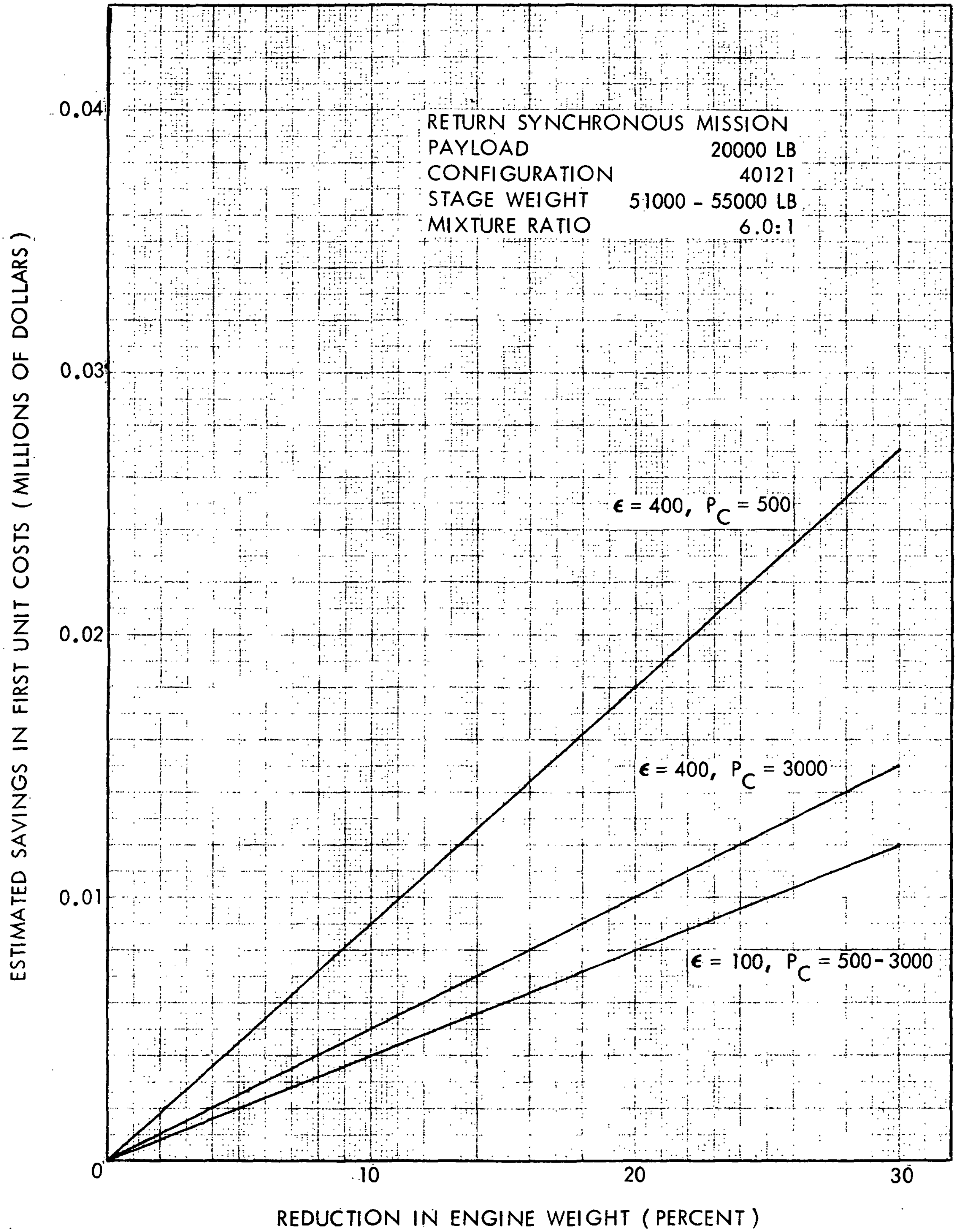

Figure 2-129. Estimated Savings in TFU Costs (51-55,000-LB Stage)

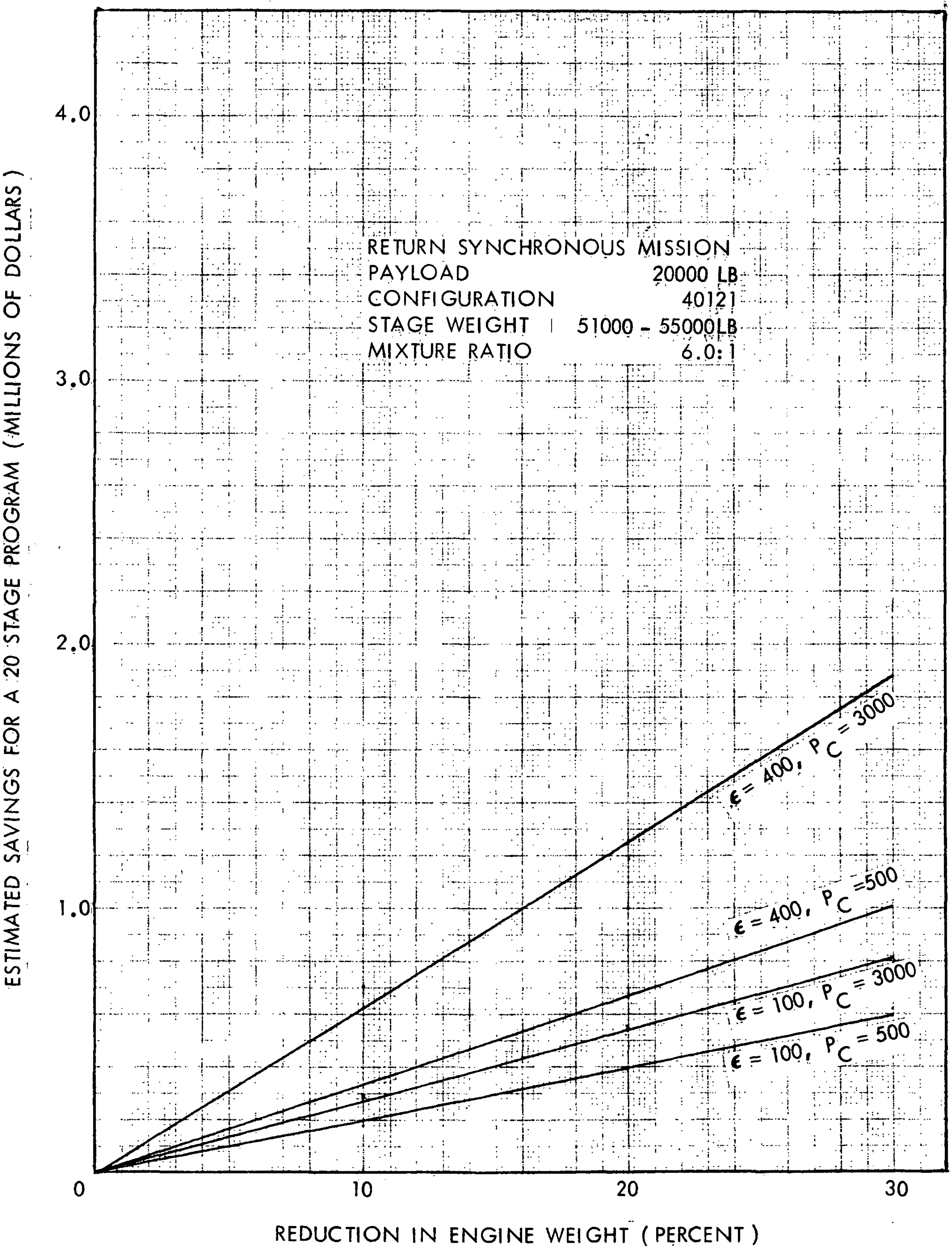

Figure 2-130. Estimated Savings for a 20-Stage Program (51-55,000 Stage)

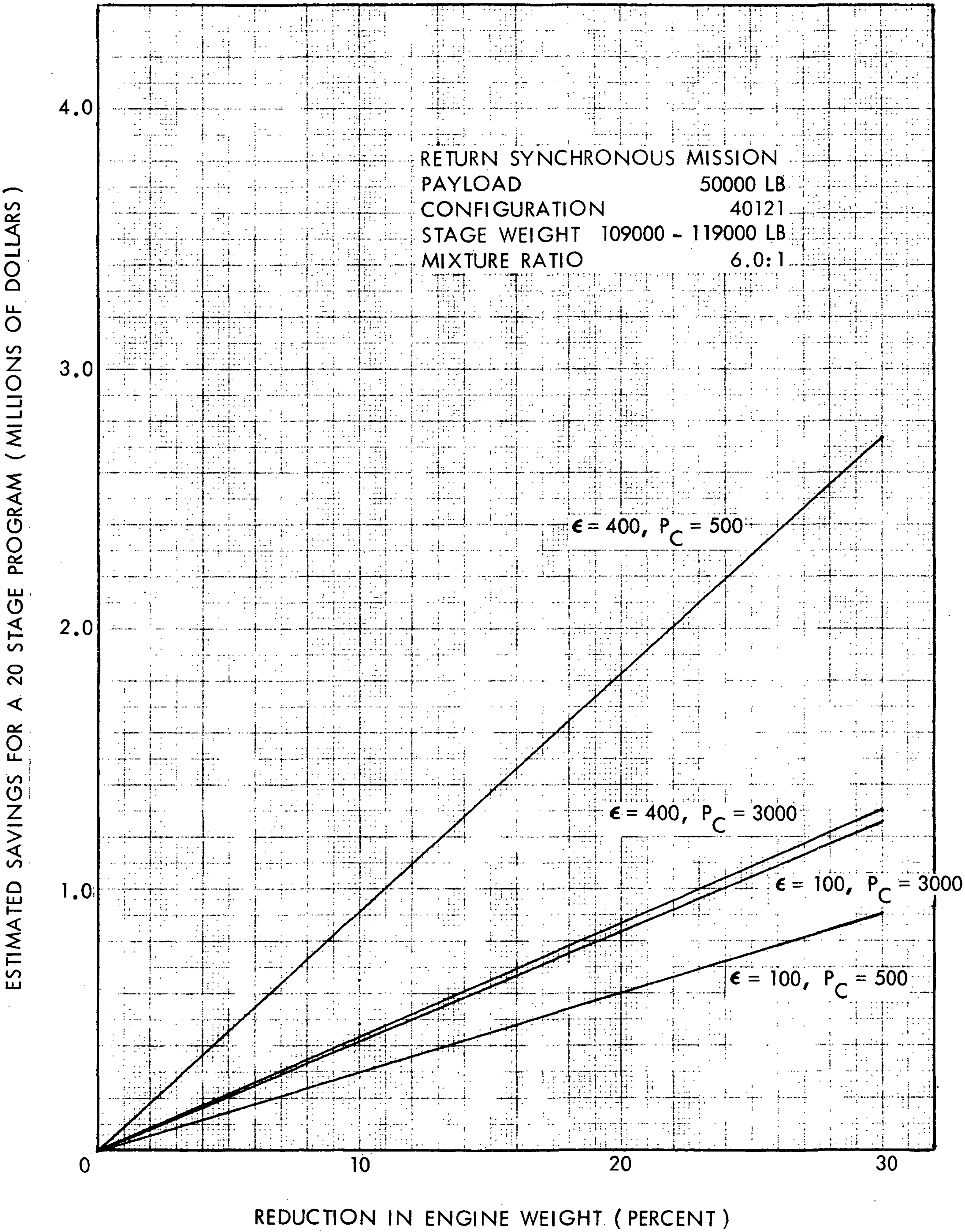

Figure 2-131. Reduction in Stage Weight (109-119,000-LB Stage)

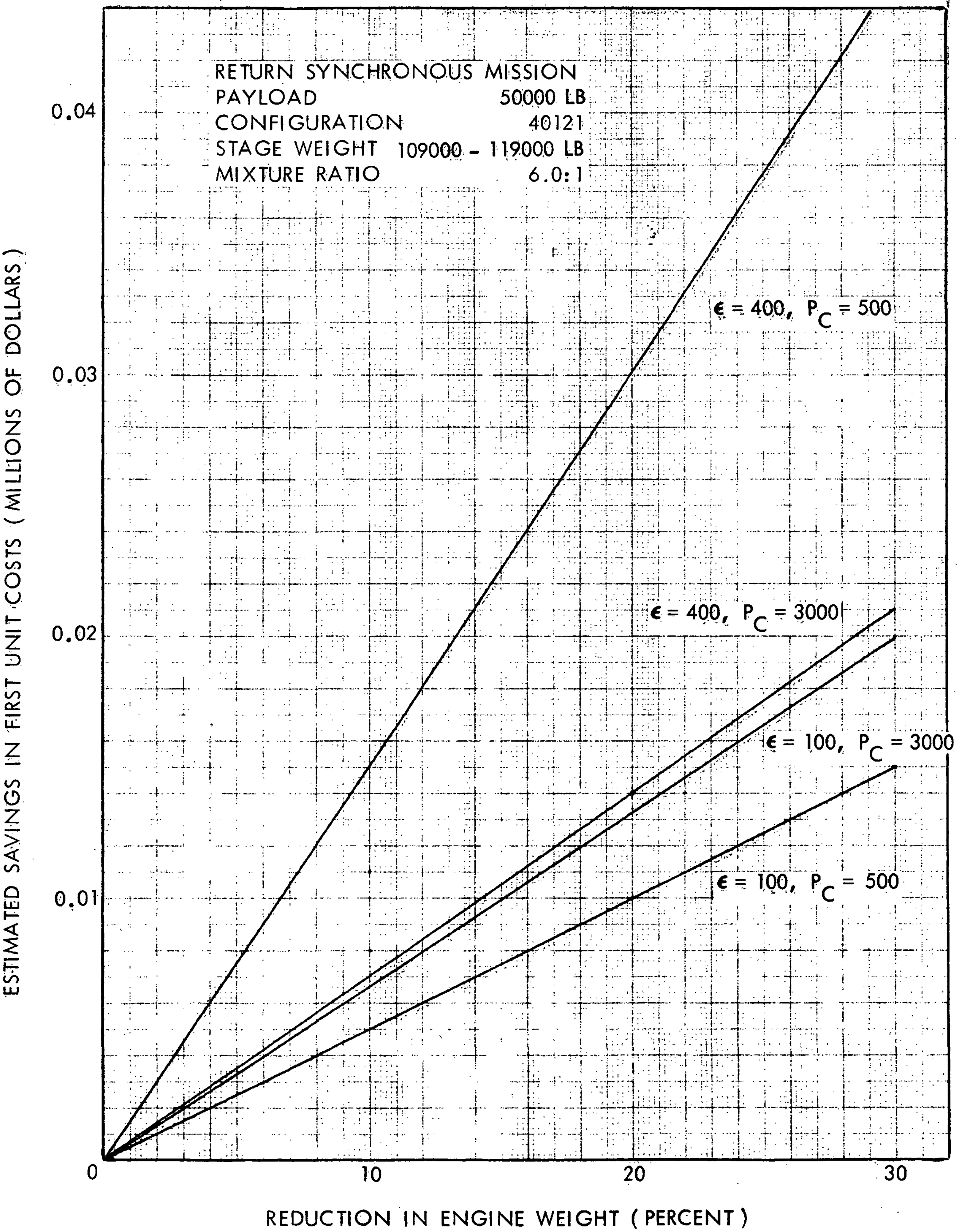

Figure 2-132. Reduction in Stage Length (109-119,000-LB Stage)

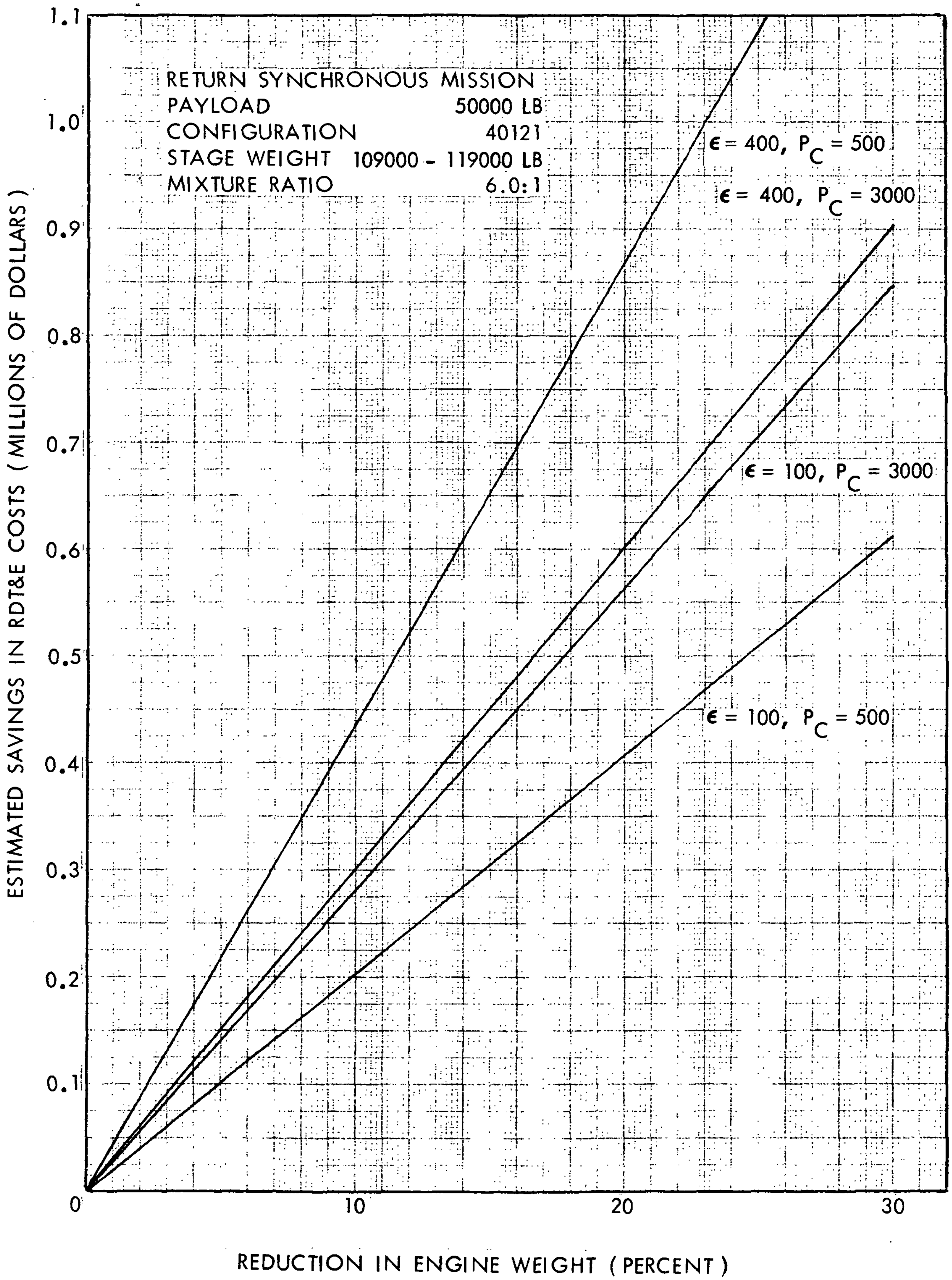

Figure 2-133. Estimated Savings in RDT&E Costs (109-119,000-LB Stage)

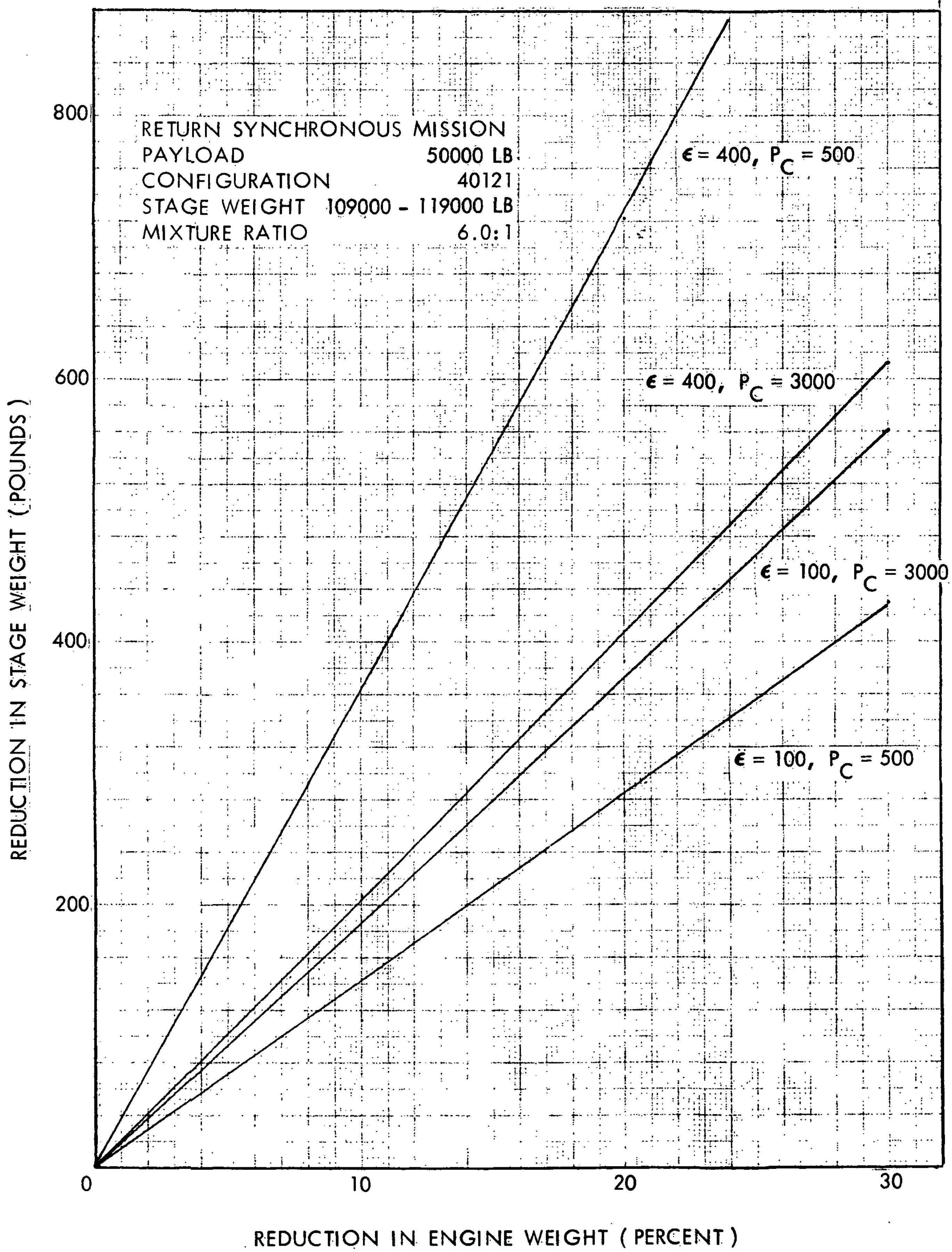

Figure 2-134. Estimated Savings in TFU Costs (109-119,000-LB Stage)

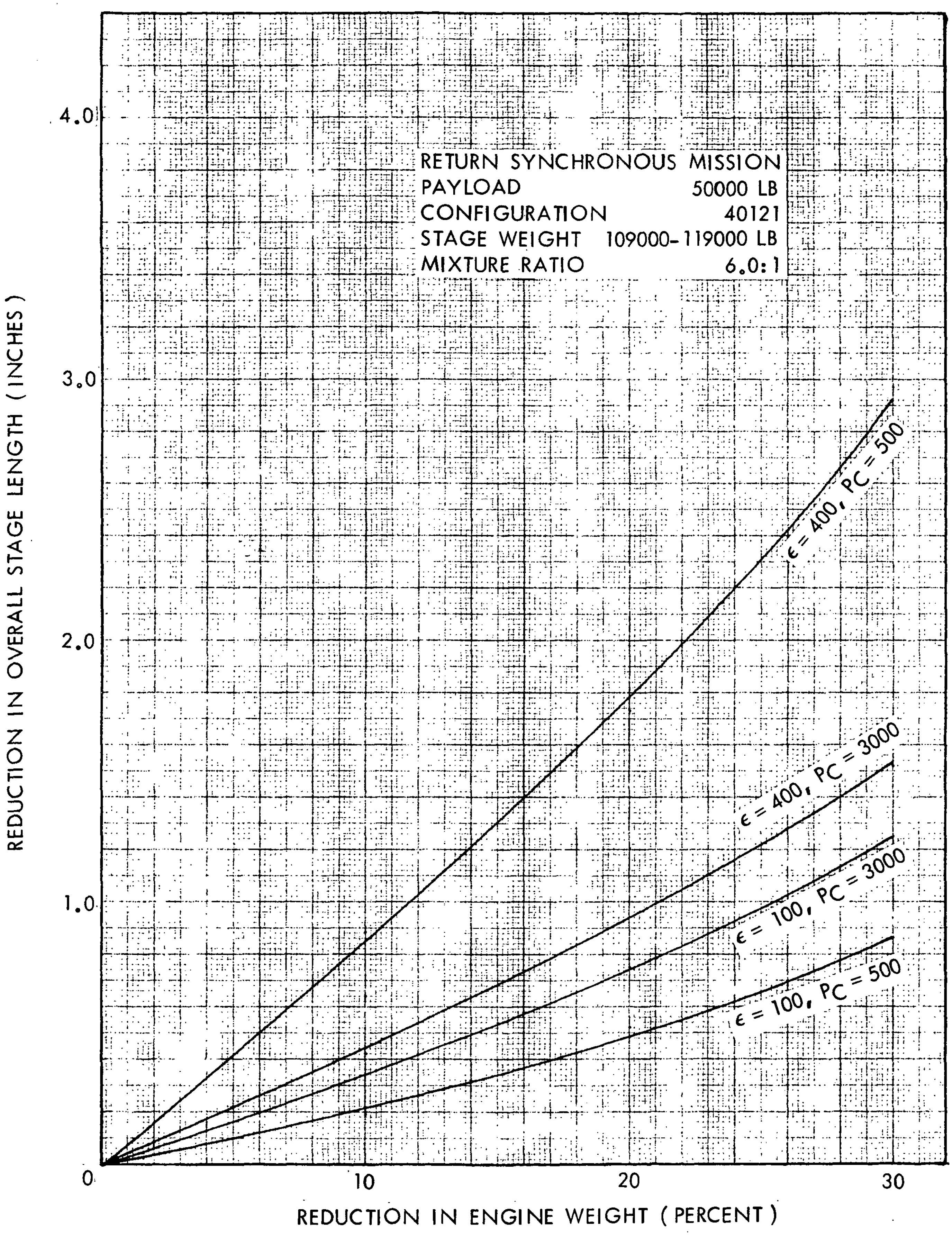

Figure 2-135. Estimated Savings in a 20-Stage Program (109-119,000-LB Stage)

Section 3

TASK 5 - WORK ORDER 5., RELATIVE GAINS OBTAINABLE THROUGH IMPROVED MASS FRACTION

3.1 GENERAL

As a result of the analyses of the effects of moderate increases in chamber pressure and engine weight reductions (subsections 2.4 and 2.5) conducted as part of Work Order 1, it was decided to replace Work Order 2 with an additional task assignment. The purpose of this task, Work Order 5, was to study the relative importance of other means of improving a stage's mass fraction. This was accomplished by varying different parameters, such as the prime structure weight, miscellaneous subsystem weights, thermal conductivity of tank insulation, etc., and determining the corresponding stage sizes and costs. The break-even costs associated with developing the technology required to obtain the various mass fraction improvements, including the effect of stage size on shuttle transportation costs, were also determined.

The results obtained in this study are discussed in the remainder of this section.

3.2 DATA AND ASSUMPTIONS

Throughout this study, certain constraints, guidelines and pertinent design data were used, and are summarized in this section. Table 3-1 gives the design constraints used and table 3-2 presents the prime structure data used in computing the shell and thrust structure weights. Table 3-3 summarizes the assumed tankage design data, including pertinent thermal protection data. The weights assumed for the astrionics systems and other miscellaneous subsystems are given in table 3-4. These weights reflect the miscellaneous subsystem philosophy recently being considered in an in-house MSFC Space Tug study (7).

The stage geometry selected as the baseline for this analysis was the 40121 configuration which has a large single hydrogen tank with ellipsoidal domes, and four small oxygen tanks with hemispherical bulkheads, suspended below the thrust cone. This stage geometry was selected instead of the tandem tank configuration because a shorter stage appears to be more advantageous for use in the shuttle's cargo bay.

The parametric oxygen-hydrogen engine system performance, weight and geometry data used in this study, were obtained from Rocketdyne for use in the "LOX/Hydrogen Engine Technology for Advanced Missions" study, contract NAS7-790. These data cover engines utilizing topping, expander and gas generator cycles. The data for the topping and expander cycles included thrust levels ranging from 15,000 to 120,000 pounds, area ratios from 100 to

Table 3-1. Summary of Stage Design Constraints

Constraint \ Mission	Single Stage Synchronous
Maximum Stage Diameter (In.)	174.0
Shell - Tank Spacing (In.)	6.0
Tank - Tank Spacing (In.)	6.0
Engine - Tank Spacing Factor (Chamber)	4.0
Engine - Tank Spacing Factor (Exit)	0.8
Engine - Booster Spacing (In.)	0.0
Engine Gimbal Angle (Degrees)	3.0
Thrust - To - Weight Ratio	0.25
Axial Acceleration (G's)	1.00
Lateral Acceleration (G's)	0.05
Payload Density (Lb/Ft^3)	25.0
Inert Weight Contingency Factor (%)	7.5

Table 3-2. Summary of Structural Design Data

Data \ Structure	Shell	Thrust Cone
Material	Aluminum	Aluminum
Density (lb/ft^3)	183.0	183.0
Material Strength (psi)		
Tension	67,000	67,000
Compression	46,000	46,000
Modulus of Elasticity (psi)	10^7	10^7
Safety Factors		
Tension	1.25	1.25
Compression	1.00	1.00
Monocoque-to-Complex Structure Weight Ratio	*	*
Spider Beam Multiplication Factor	N/A	N/A

* A function of diameter and limit load; see appendix C.

Table 3-3. Summary of Tankage Design Data

Data \ Mission	Synchronous
Tankage	
Material	Aluminum
Density (lb/ft^3)	183.0
Allowable Stress (psi)	60,000
Factor of Safety	1.10
Minimum Skin Gauge (In.)	0.025
Land Factors (Bulkheads)	0.10
Land Factors (Cylindrical Section)	0.05
Thermal Protection	
Initial Fuel/Oxidizer Temperature (°R)	36.0/162.6
Initial Fuel/Oxidizer Pressure (psi)	15.0/15.0
External Insulation Temperature (°R)	450.0/470.0
Insulation Density (lb/ft^3)	4.5
Insulation Thermal Conductivity (Btu/Hr-Ft-°R)	*
Meteoroid Protection	
Probability of no Punctures	N/A
Nominal Mission Altitude (n.m.)	N/A
Shield Material	N/A
Material Density (lb/ft^3)	N/A
Material Yield Stress (psi)	N/A
Minimum Skin Gauges (In.)	N/A
Miscellaneous	
Minimum Fuel/Oxidizer Ullage Volume (%)	5.0/5.0
Residual Fuel/Oxidizer Fraction (%)	2.0/2.0
Feedline Flow Velocity (fps)	20.0
Tank Support Factor	ξ

* A function of temperature and thickness; see appendix C.

ξ Dependent upon configuration; see appendix C.

Table 3-4. Miscellaneous Subsystem Weights (MSFC TUG)

System	Weight
Electric Distribution	200 lb
Electric Power	300
Power Systems	
Fuel Cell Reactants	
Communication/Instrumentation	295
Communications	
Data Management	
Instrumentation	
Guidance, Navigation, and Control	210
Guidance, etc.	
Rendezvous and Docking Radar	
Hydraulic/Pneumatic	145
Purge	
Umbilical	
Tug/Orbiter Service	
Propellant Utilization	35
Miscellaneous	715
Destruct System	
Docking Adapter	
Subsystem Mounts	
Orbiter Interface	
Payload Interface	
TOTAL	1900 lb

400, and mixture ratios of 5.0, 6.0 and 7.0. Data were supplied for chamber pressures of 500 to 1000 psi, and 1000 to 3000 psi, for the expander and topping cycles, respectively. The gas generator cycle data covered lower thrust engines (5,000 to 15,000 lb) and chamber pressures of 800 and 1000 psi.[1] These parametric engine data were published in appendix B of the final report for the study cited above.

In computing the break-even costs associated with the various improvements in technology, it was necessary to assess the implications which changes in stage length would have on shuttle transportation costs. The variations in shuttle payload capability (stage plus extra cargo) were determined from the shuttle cargo bay criteria depicted in table 3-5.

Table 3-5. Shuttle Cargo Bay Criteria

Cargo Bay Length	60.0 ft
Cargo Bay Diameter	15.0 ft
Maximum Allowable Cargo Diameter	14.5 ft
Maximum Allowable Cargo Volume	9908 ft^3
Maximum Allowable Cargo Weight	65,000 lb

The density of the extra payloads used in this analysis was computed on the basis of maximum allowable shuttle cargo weight and carbo bay volume. This density, 6.56 lb/ft^3, is slightly greater than the average density of approximately seventy shuttle payloads considered in a recent study.[8] Both of these densities are considerably larger than the density (2 lb/ft^3) which would be required to fill the remainder of the cargo bay not occupied by the stage, while maintaining the maximum cargo weight limitation of the shuttle. This results from the large 64,000-pound stages being considered in this particular analysis. However, if small 10-15,000-pound stages were being considered, the density of the extra cargo required to fill remaining cargo bay area would be approximately 10 lb/ft^3.

3.3 MISSION PROFILE

The mission profile selected for this analysis was the round-trip transfer of a payload between a low-inclination, low-altitude earth (parking) orbit and a synchronous orbit. This mission would require the liquid hydrogen-liquid oxygen stage to transport a 3000-pound payload from the low-parking orbit to synchronous orbit and return with the same or a different 3000-pound payload.

The typical profile for this mission is depicted in figure 3-1. This involves a Hohmann-type transfer maneuver from low earth orbit to synchronous altitude, a plane change and circularization at synchronous orbit, a return Hohmann-type transfer and plane change at synchronous orbit, and circularization into the original low earth orbit.

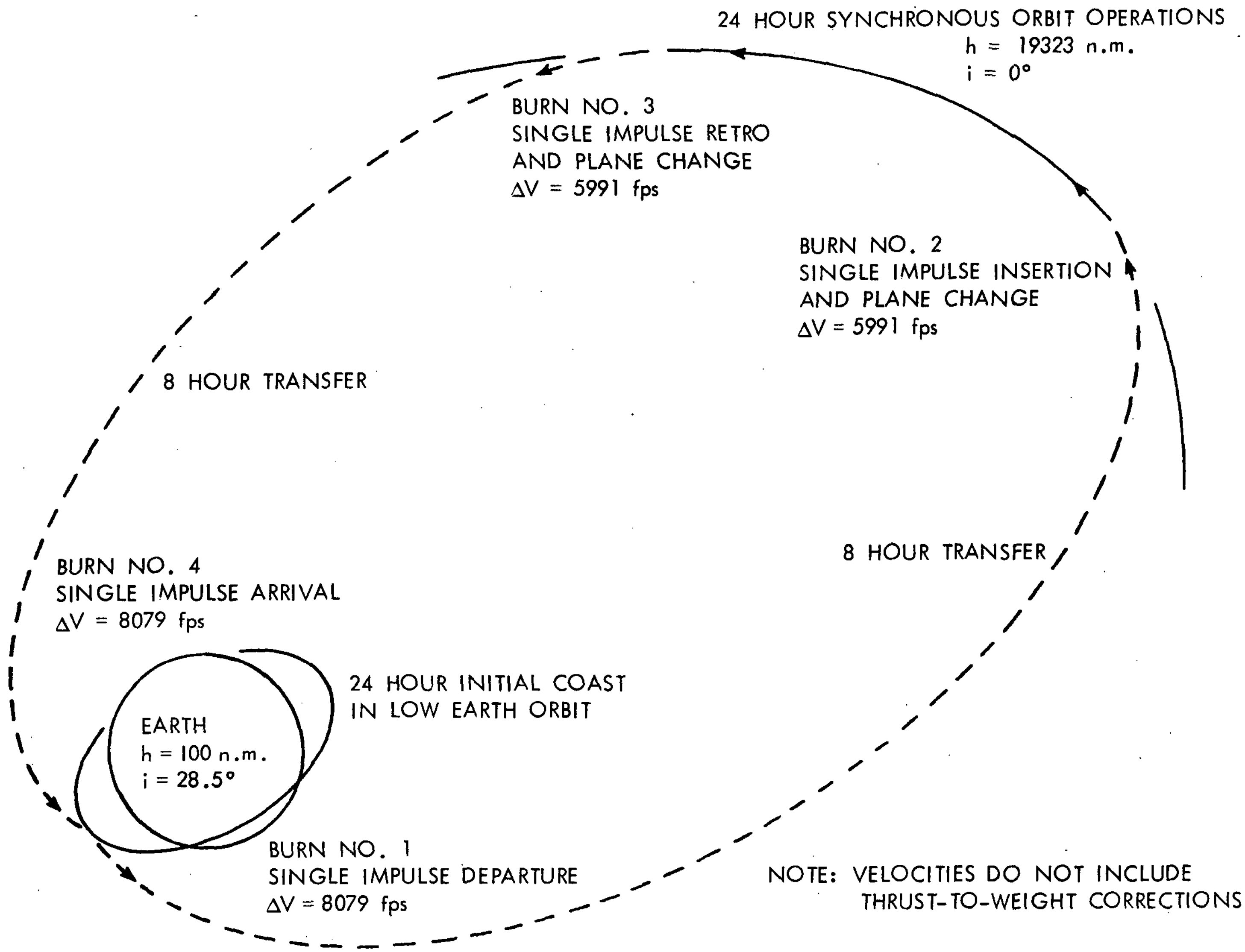

Figure 3-1. Synchronous Mission Profile

The four velocities used for this mission assumed a Hohmann-type transfer between a 28 ½-degree inclination, 100-nautical-mile circular orbit and an equatorial (0-degree inclination) synchronous orbit. The velocities were corrected to account for the effects of the stage's initial thrust-to-weight ratio and specific impulse. However, the effect of orbital regression on the velocity requirements was not considered.

3.4 BASELINE STAGE

The selection of a baseline stage for this analysis was made on the basis of the maximum shuttle cargo capability (65,000 pounds). Table 3-6 presents a weight statement for the baseline stage which was the largest stage sized that satisfied the above weight limitation and could still carry a 3000-pound payload on the round-trip synchronous mission. This baseline stage weighs 64,443 pounds at ignition and has a burn-out weight of 7169 pounds, and exhibits a propellant fraction of 0.884. Table 3-7 summarizes some of the more salient features of this stage; while its external profile is illustrated in figure 3-2. A detailed cost summary is presented in table 3-8 for a 20-stage program.

The costs generated during this analysis were based on cost estimating relationships which are predicated on historical cost data and pertinent vehicle parameters.(4) In general, the cost estimating relationships of any cost element contain coefficients which indicate the technology level and complexity of that individual element. Table 3-9 presents the baseline technology levels assumed for the main systems on the stage and table 3-10 lists the percent learning curves used to compute the investment costs.

As indicated in the cost summary shown in table 3-8, the program cost data developed during this analysis include only the RDT&E, investment, and upper stage propellant costs. The program costs presented do not contain the other cost elements normally included in the operations costs, because they are mainly launch vehicle and mission model orientated, and for identical missions and similarly sized upper stages the operational costs are relatively insensitive to variations in upper stage weight. Hence, any program cost sensitivities which are determined from the program cost data will be of sufficient accuracy.

3.5 ENGINE PARAMETERS

Because the baseline mission and stage selected for this study differed from those used in analyses of moderate chamber pressure and engine weight conducted under Work Order 1, it was necessary to reevaluate the influence of chamber pressure and engine weight. The results of these two analyses, and an analysis of the effect of the engine nozzle expansion ratio are presented in the remainder of this subsection.

3.5.1 The Effect of Chamber Pressure

The influence of moderate increases in chamber pressure (750 to 3000 psi) is depicted in figures 3-3 through 3-10. Variations in stage size are presented in figures 3-3, 3-4 and 3-5, which show the changes in stage weight, length and volume, respectively. These data show that although an average change in stage weight results from increased chamber pressure, a large variation in stage length is obtained. This is mainly due to the decreases

Table 3-6. Weight Summary for the Baseline Stage

STRUCTURE			2306
TANKAGE		723	
Hydrogen	436		
Oxygen	287		
INTERSTAGE		0	
LANDING SYSTEMS		0	
SHELL		466	
THRUST STRUCTURE		154	
OTHER(TANK SUPPORTS,FEED SYSTEMS)		963	
METEOROID SHIELD			0
INSULATION			187
FUEL TANK		124	
OXIDIZER TANK		63	
PROPULSION			686
ENGINE		385	
OTHER INERT(RCS,PRESS)		301	
ASTRIONICS			1005
MISCELLANEOUS FIXED WEIGHTS			895
CONTINGENCY(@ 7.5 %)			508

PROPELLANT INVENTORY			58795
TOTAL FUEL LOAD		8522	
Useable	8135		
Residual	170		
Vented	0		
Final Ullage	176		
Startup/Shutdown	41		
TOTAL OXIDIZER LOAD		50273	
Useable	48812		
Residual	1006		
Vented	0		
Final Ullage	209		
Startup/Shutdown	246		
PRESSURIZATION & INERT GASES			21
TOTAL LOAD		21	
Vented	0		
Residual	21		
RCS PROPELLANT			40
TOTAL FLUIDS			58856
CONSUMED		57274	
VENTED		0	
RESIDUAL		1582	

BURN OUT WEIGHT	7169
TOTAL STAGE WEIGHT	64443
PROPELLANT FRACTION	0.8837

Table 3-7. Design Data Summary for the Baseline Stage

COAST AND BURN NUMBER	1	2	3	4
COAST TIME (hours)	14	6	60	6
PAYLOAD (lb)	3000	3000	3000	3000
VELOCITY INCREMENT(TOTAL= 28,346 fps)	8222	5951	5951	8222
PROPELLANT BURNED (lb)	28273	12711	8553	7410
FUEL VENT(incl. Press/Inert Gases)	0	0	0	0
OXID. VENT(incl. Press/Inert Gases)	0	0	0	0

ENGINE CHARACTERISTICS	
Thrust (lb)	16860
Specific Impulse (sec)	470.1
Area Ratio	400:1
Mixture Ratio	6.0:1
Chamber Pressure	750

TANKAGE	FUEL	OXIDIZER
Vent Pressure (psi)	23.50	36.00
Design Pressure (psi)	26.98	39.60
Insulation Thickness (in.)	0.42	0.24
Additional Meteoroid Shield Thickness		
Bumper	N/A	N/A
Backup Wall	N/A	N/A
Volume	2011.6	185.8

STAGE GEOMETRY	
Overall Length (in.)	406.1
Maximum Diameter (in.)	174.0
Stage L/D	2.33

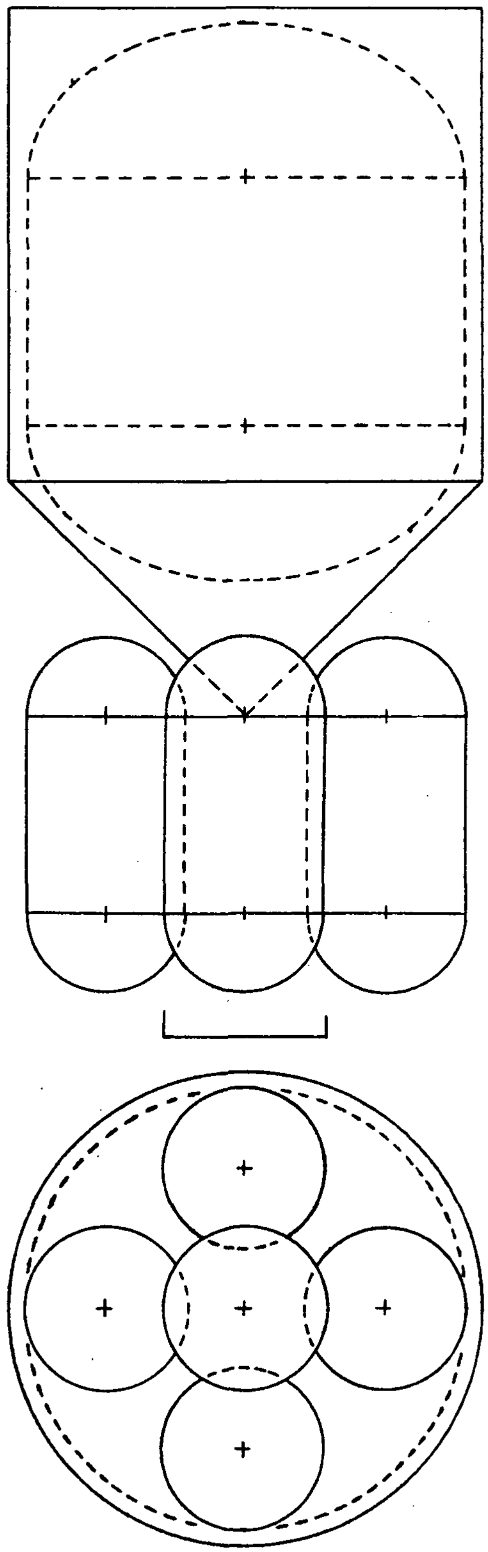

Figure 3-2. Baseline Stage Configuration

Table 3-8. Cost Summary for the Baseline Stage (20-Stage Program) *

Item/Area	RDT&E	Investment	Operations	Total	Percent	First Unit
Structure	130.020	97.431		227.452	22.01	6.670
Propulsion	188.825	80.255		269.081	26.04	5.540
Miscellaneous Systems	74.194	69.163		143.358	13.87	4.735
Facilities-Tooling-Equipment	62.138	3.321		65.459	6.33	
Training	1.500		.000	1.500	.15	
Ground Support Equipment	13.796	9.657		23.454	2.27	
Ground Test	45.345			45.345	4.39	
Flight Test	112.399			112.399	10.88	
System Integration	23.438			23.438	2.27	
Project Management	32.547	1.413		33.960	3.29	
Assembly-Checkout		10.484		10.484	1.01	.718
Initial Operating Spares		25.733	.000	25.733	2.49	
Sustaining Engineering		51.467	.000	51.467	4.98	
Launch Crew			.000	.000	.00	
Propellants			.180	.180	.02	
Payload Integration			.000	.000	.00	
Equipment Maintenance			.000	.000	.00	
Booster or Launch Vehicle			.000	.000	.00	
Stage Recovery			.000	.000	.00	
Stage Refurbishment (Ground)			.000	.000	.00	
Orbital Operations			.000	.000	.00	
Maintenance of Capability			.000	.000	.00	
Total Cost of Program	684.203	348.925	.180	1033.308	100.00	17.662
Percent of Total Program	66.21	33.77	.02	100.00		

* All costs are in millions of dollars

Table 3-9. Technology Level of Systems

Area	Technology Level and Technique *
Structures	
Shell	SOA - Aluminum Sheet Stringer
Thrust Structure	SOA - Aluminum Sheet Stringer
Tankage	SOA - Aluminum Monocoque
Meteoroid Shield	SOA - Aluminum Monocoque
Tank Supports	ADV - Composite
Propellant Feedlines	SOA - Aluminum
Propulsion	
Main Engines	New, advance, reuseable LH_2/LOX
Reaction Control Thrusters	SOA - Monopropellant
Miscellaneous Subsystems	Adaptation of existing hardware to a new unmanned, reuseable upper stage
Electrical Power and Distribution	
Electrical	
Communication	
Instrumentation	
Guidance, Navigation and Control	
Hydraulic/Pneumatic	
Propellant Utilization	
Destruct	

*SOA - State-of-the-art Technology
ADV - Advanced Technology

Table 3-10. Investment Learning Curves

System	Learning Curve
Structures	90%
Propulsion	95%
Miscellaneous Subsystems	90%
Assembly & Checkout	90%

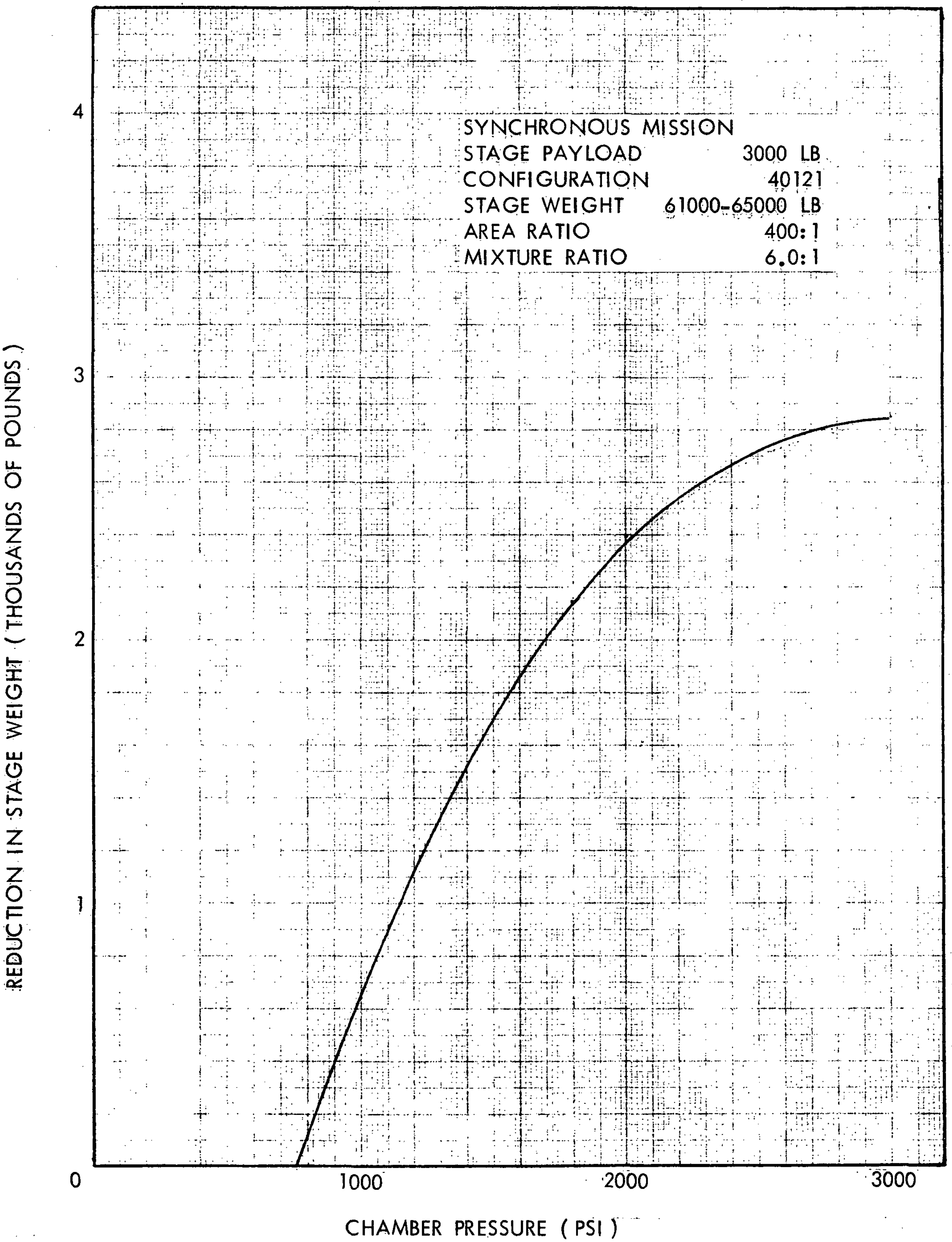

Figure 3-3. Variation of Stage Weight Due to Chamber Pressure

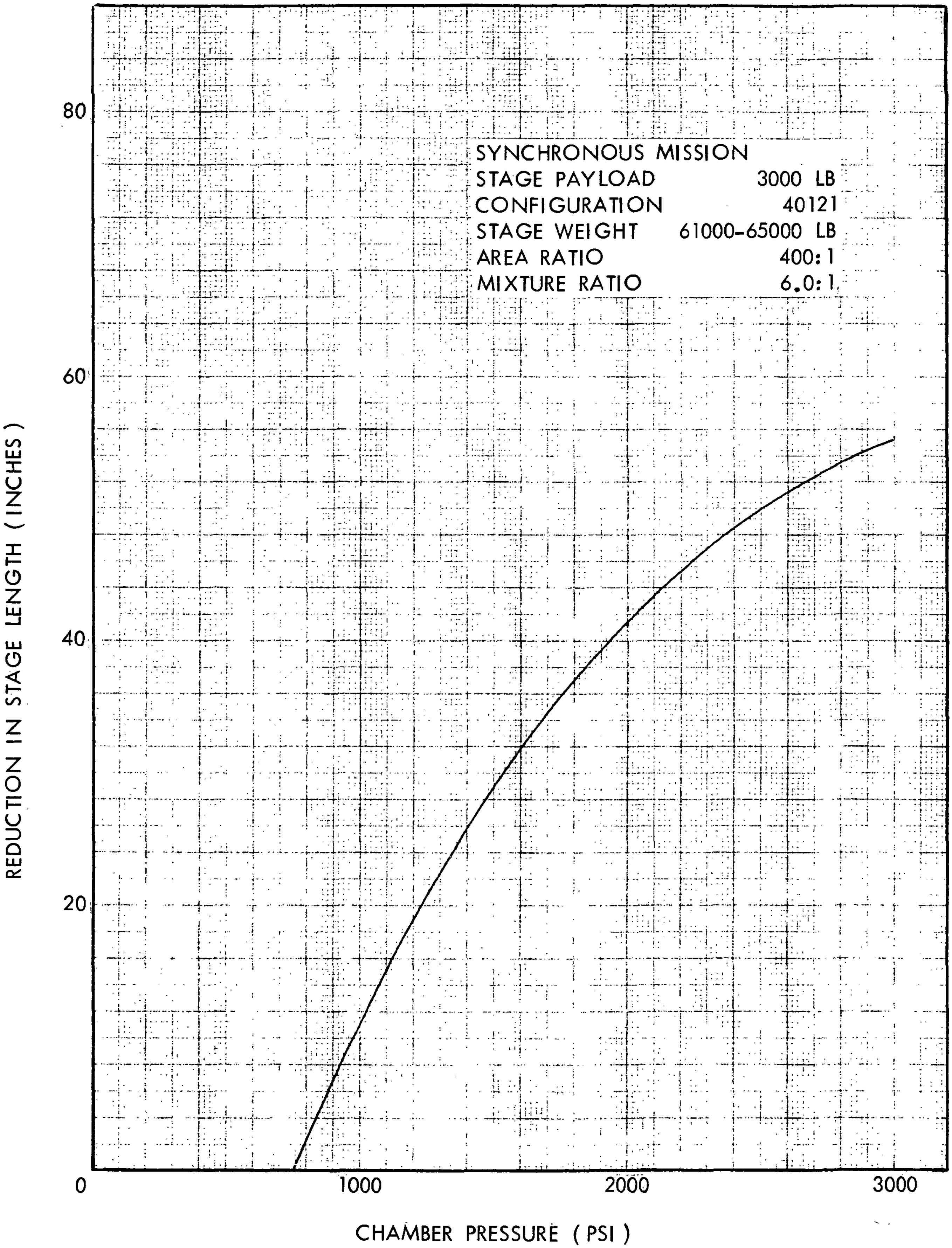

Figure 3-4. Variation of Stage Length Due to Chamber Pressure

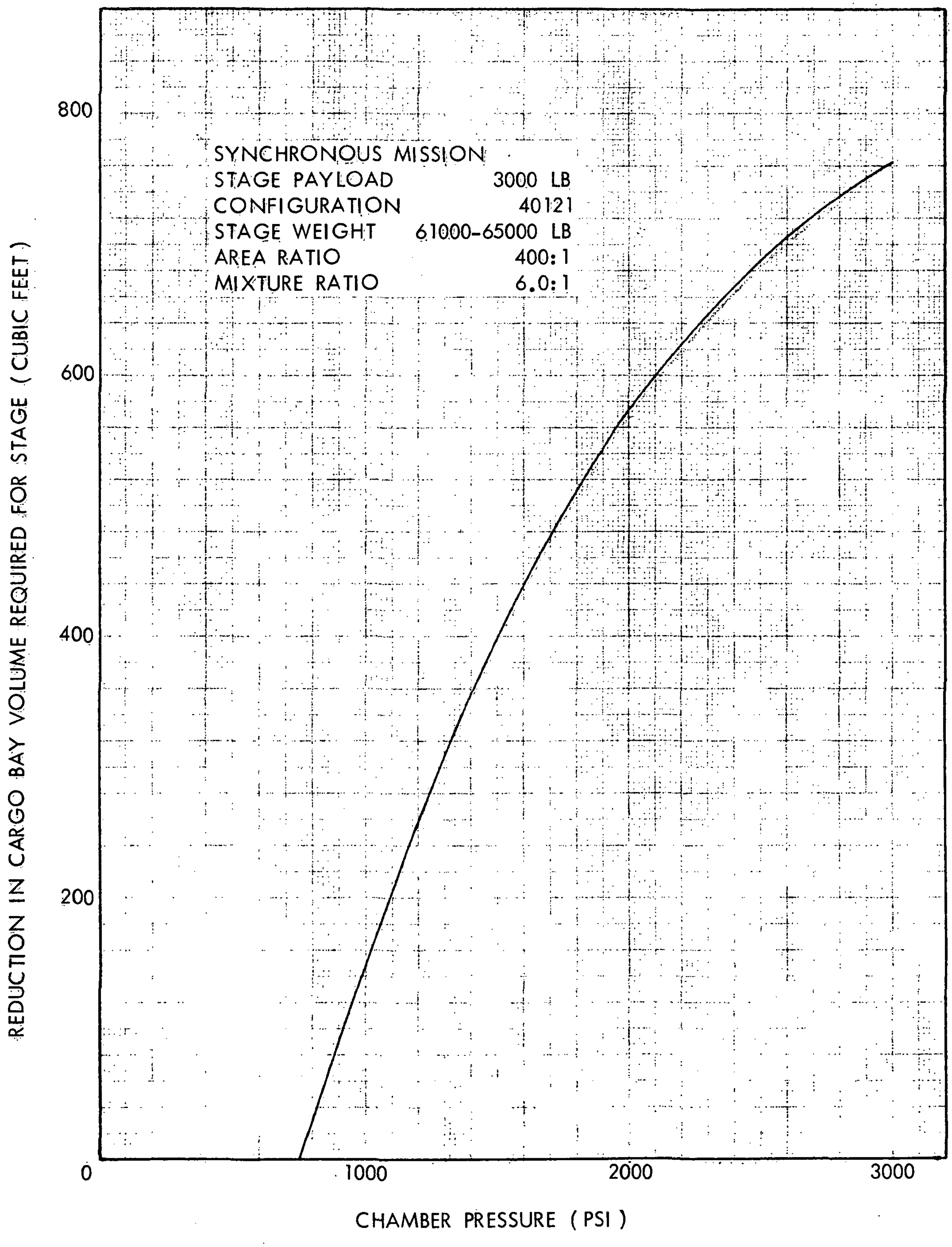

Figure 3-5. Variation of Stage Volume Due to Chamber Pressure

in engine length. This fact becomes important in the case of a volume-limiting payload being launched with the stage in the shuttle's cargo bay.

The effect of increased chamber pressure on RDT&E, TFU and program costs is depicted in figures 3-6, 3-7 and 3-8, respectively. The savings presented in these figures do not reflect the cost of the technology required to obtain the higher chamber pressures.

In order to evaluate the influence of both decreases in stage weight and length due to higher chamber pressures, it was necessary to determine the number of shuttle launches needed before the additional extra cargo gained in each launch of a stage (because of lighter, smaller stages) would equal the equivalent payload of a single shuttle launch without a stage.

The results, which show the number of stage launches required to save one shuttle flight, are presented in figure 3-9. These data are representative of a weight-limited case. That is, although the total shuttle cargo weight is 65,000 pounds (stage plus extra payload), the extra payload does not completely fill the bay volume not occupied by the stage. For the large stage being considered in this analysis, this case is more realistic than the volume-limited case.

The launch cost savings were computed from data on the number of stage launches required to save one shuttle flight, and used in conjunction with the stage savings data (figures 3-6 and 3-8) to determine the technology break-even points.

The break-even technology investment costs (technology costs = total savings) for programs having various numbers of stages are depicted in figure 3-10. These data are presented in a slightly different form then previously shown in the analysis of moderate chamber pressure increases (subsection 2.4), so that a comparison of the relative merits of the various technologies can be made.

3.5.2 The Effect of Engine Weight

A set of data, similar to that prepared for chamber pressure, which indicates the influence of engine weight reduction on stage size and cost, is presented in figures 3-11 through 3-18.

These data show that stage size and cost are less sensitive to reductions in engine weight than to increased chamber pressure, because large reductions in engine weight decrease the total inert weight of the stage only slightly, while not improving the engine's performance. Because the variations in stage weight and cost are small, the break-even technology investment costs are less than those found for the chamber pressures.

3.5.3 The Effect of Nozzle Area Ratio

Although the effect of engine nozzle expansion ratio was not originally intended to be analyzed, toward the end of the study it was decided to investigate its influence. Because the baseline stage had an area ratio of 400:1, it was necessary to define a new reference stage in order to obtain positive trends. A new stage having the same 3000-pound round-trip payload capability,

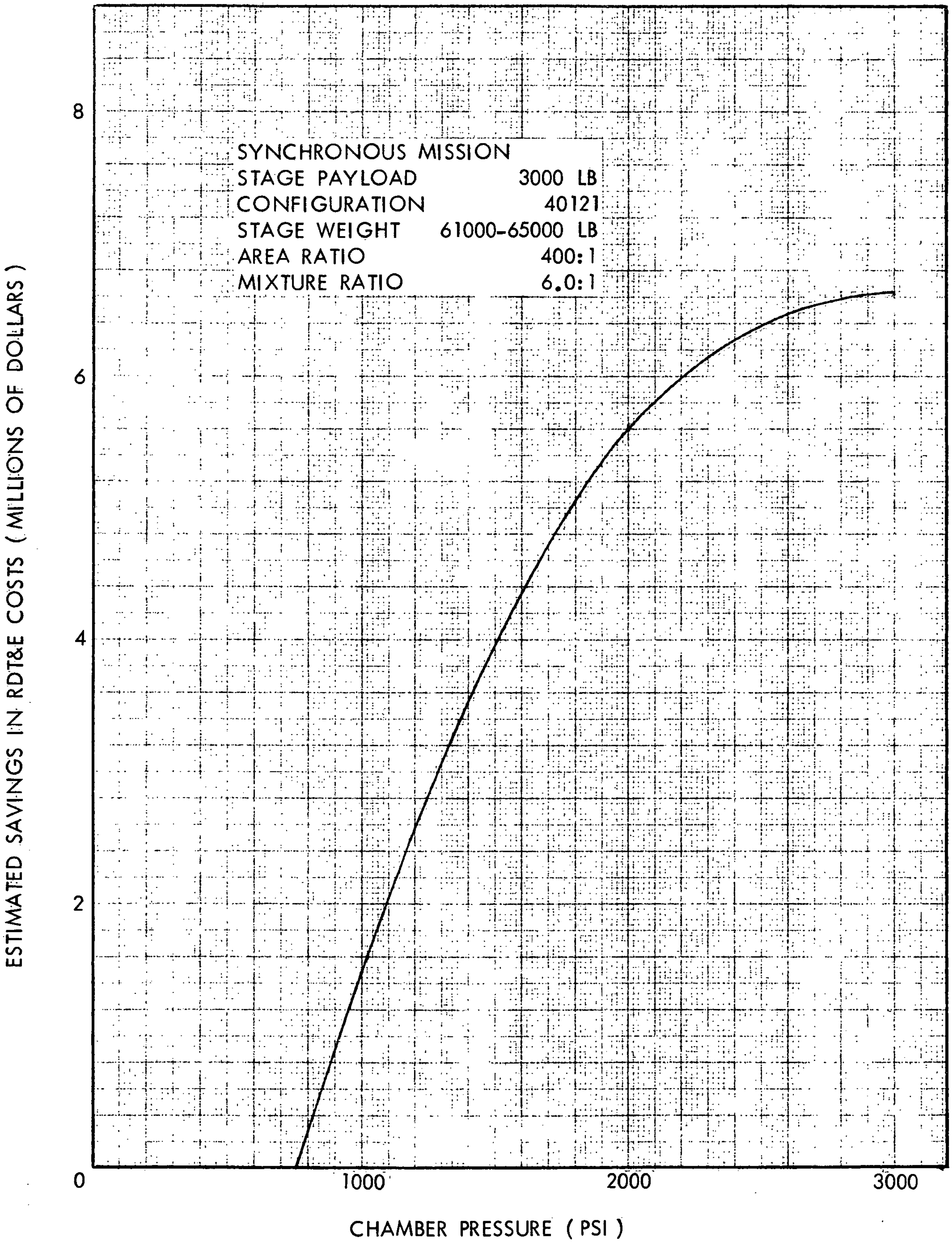

Figure 3-6. Variations of RDT&E Cost Due to Chamber Pressure

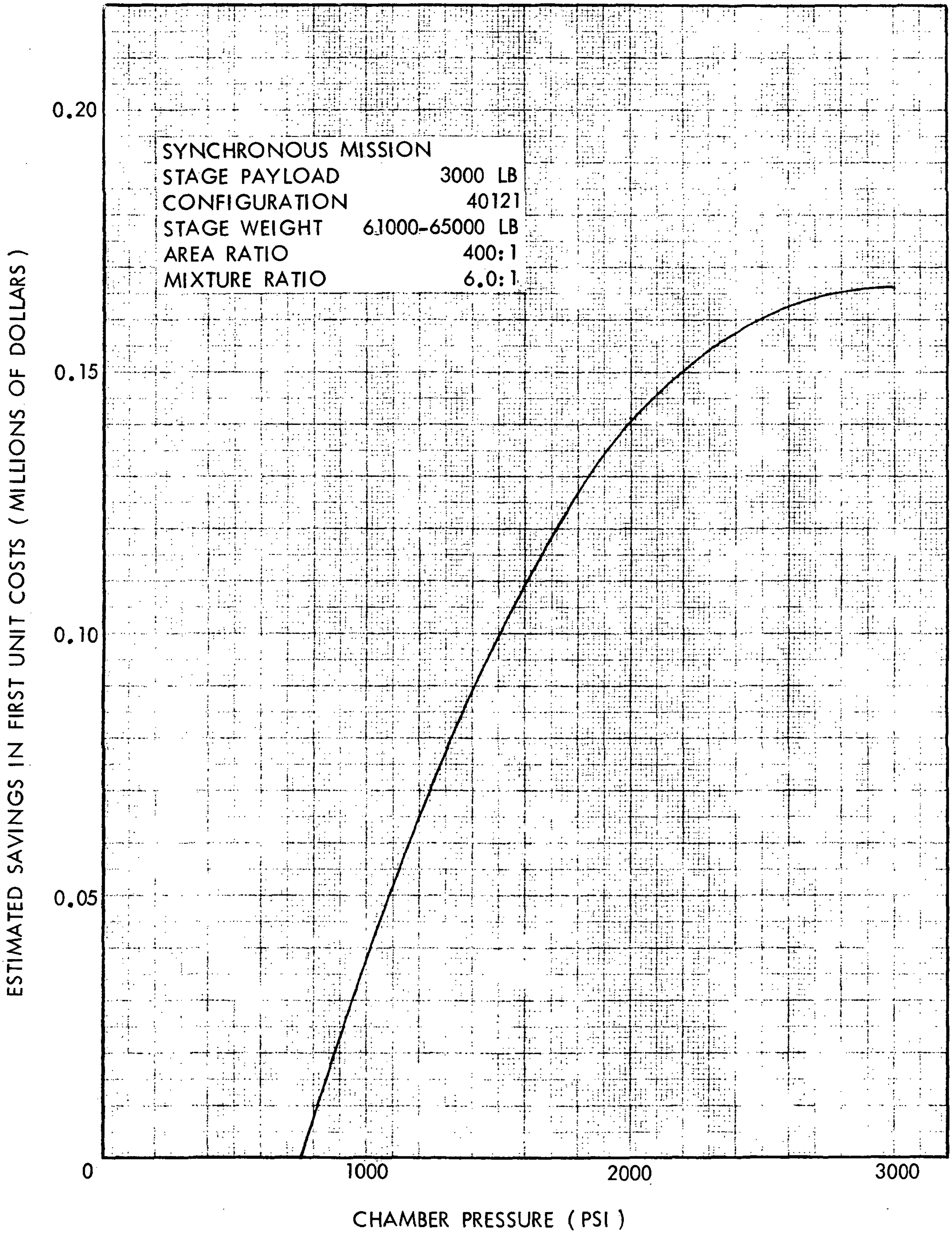

Figure 3-7. Variations of First-Unit Costs Due to Chamber Pressure

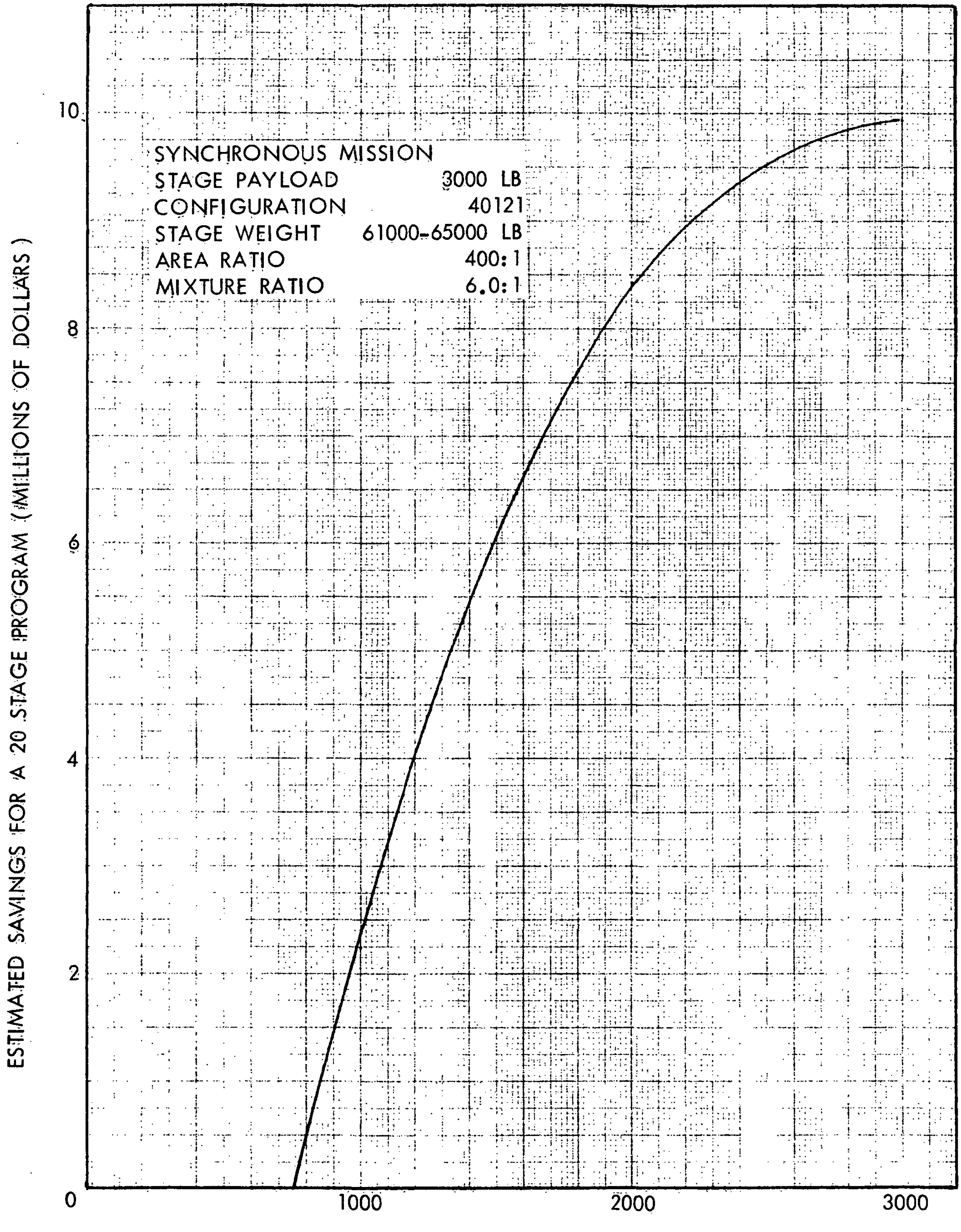

Figure 3-8. Variations of Program Cost Due to Chamber Pressure

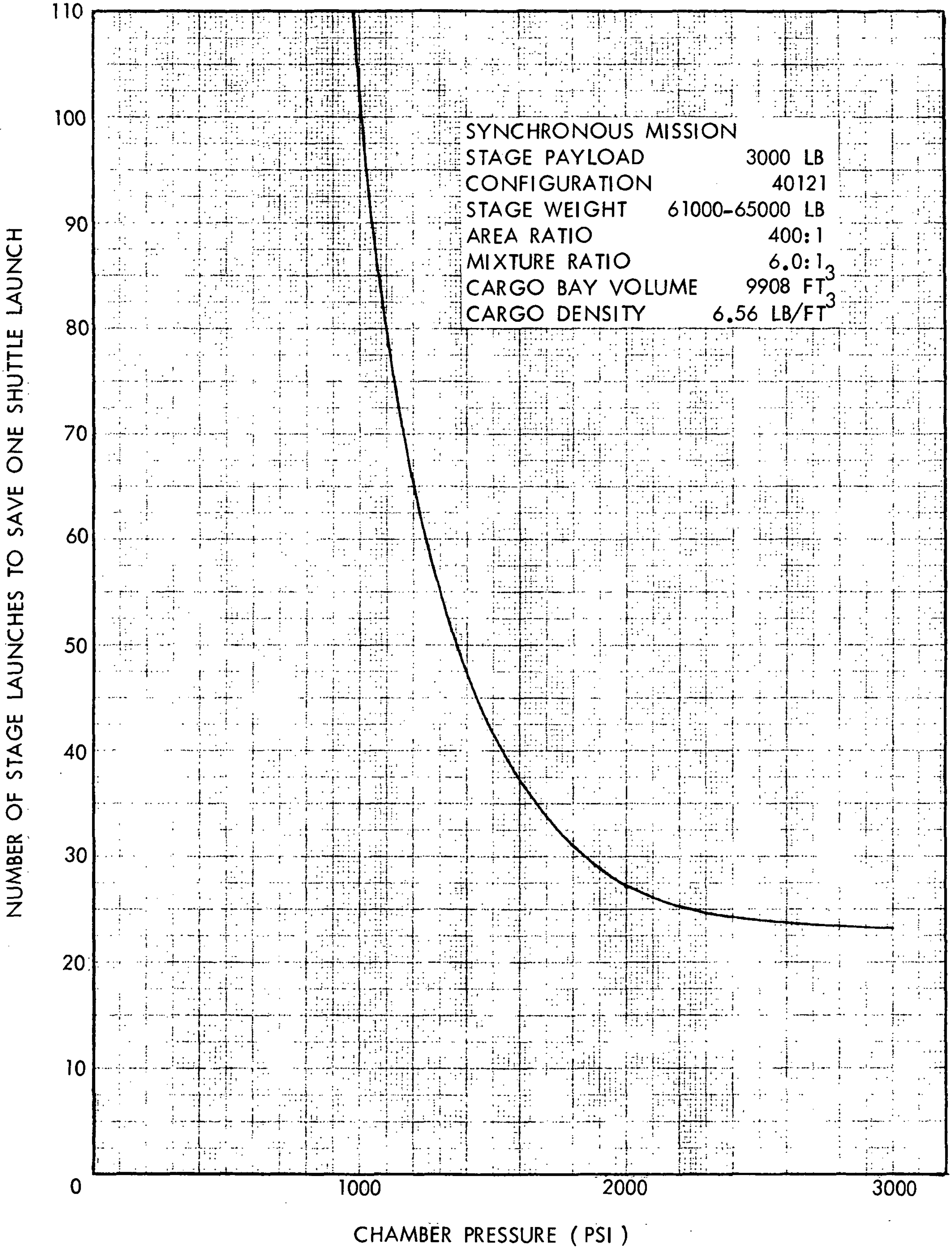

Figure 3-9. The Number of Stages Launches Required to Save One Shuttle Flight (Chamber Pressure)

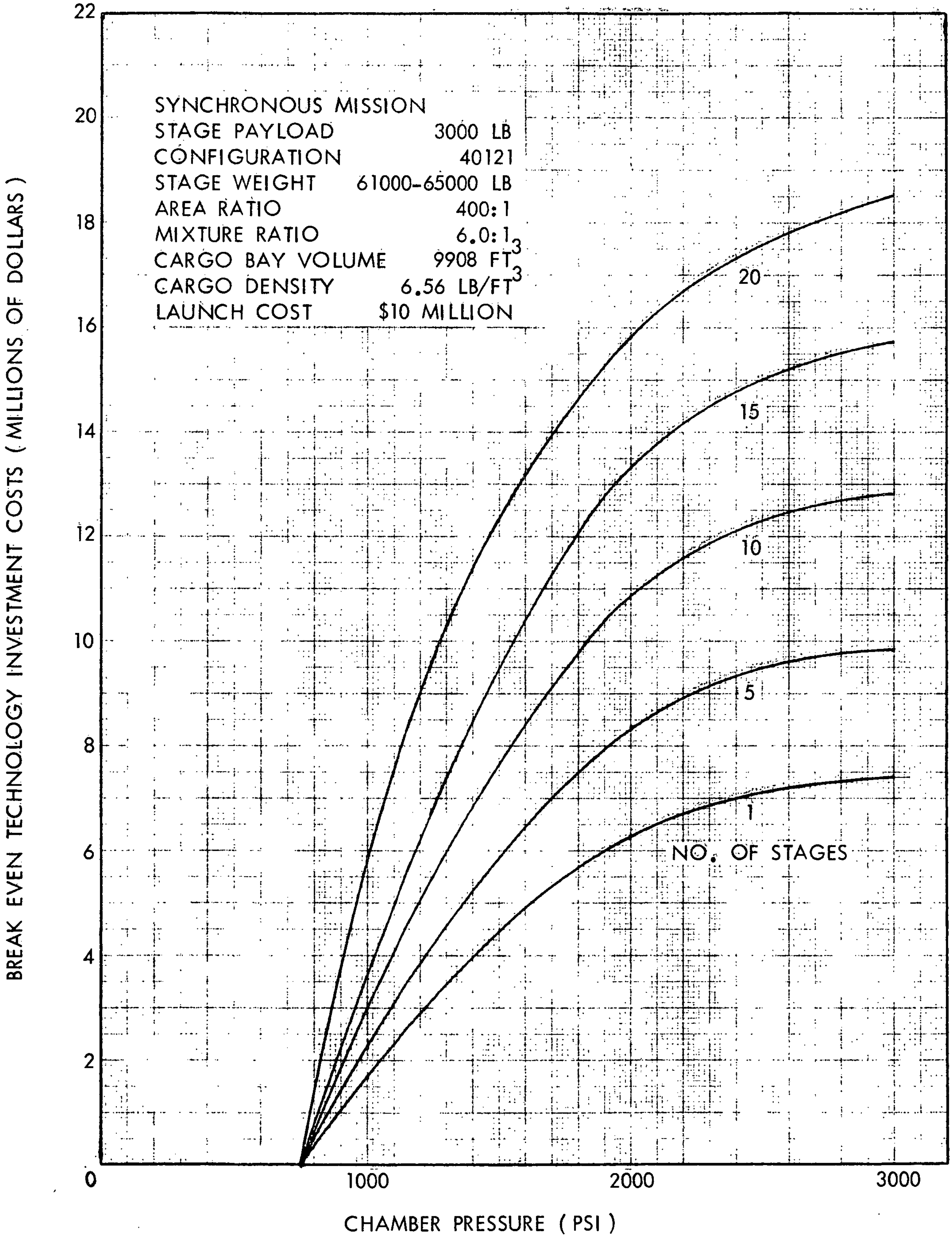

Figure 3-10. Break Even Technology Costs for Chamber Pressure

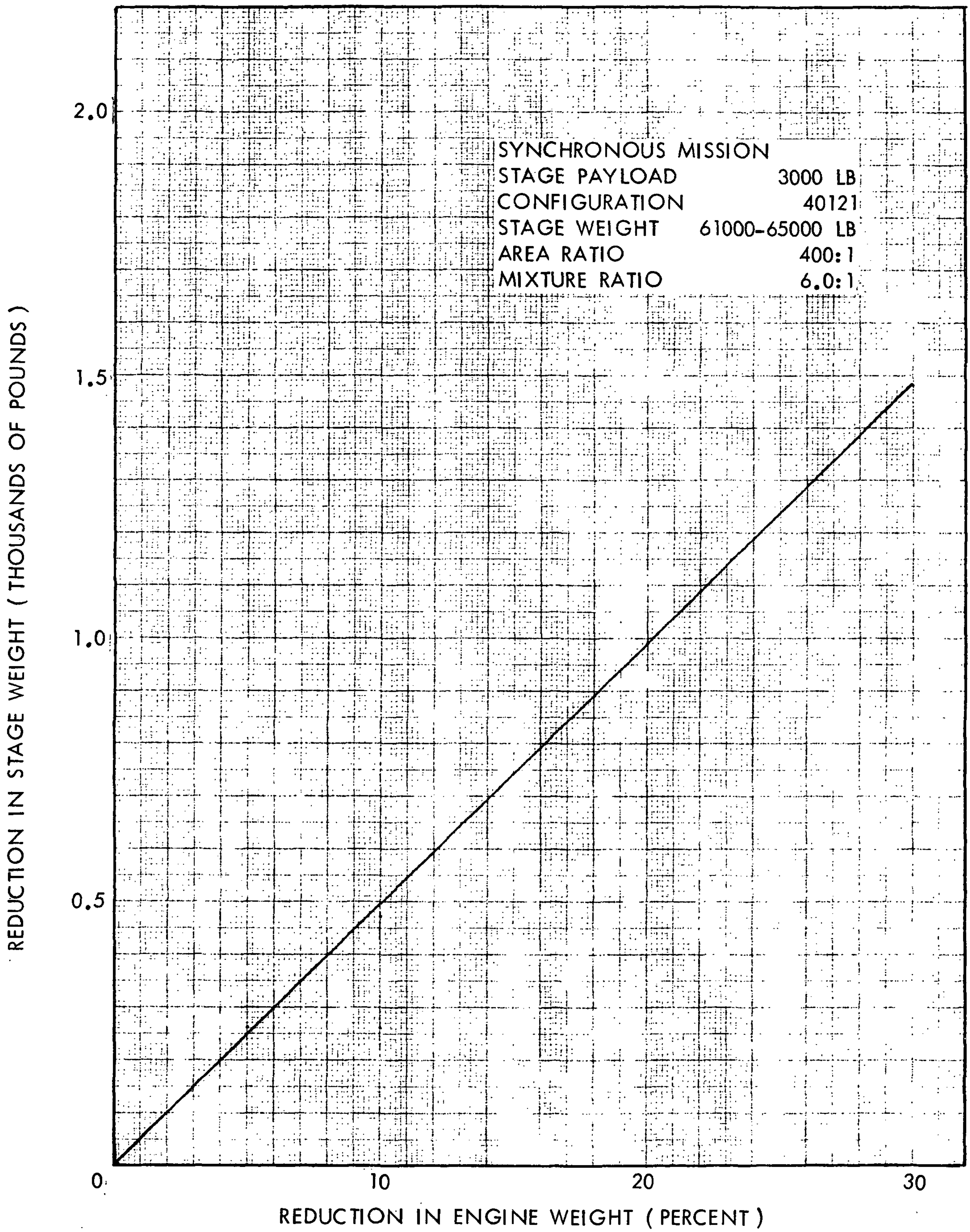

Figure 3-11. Variation of Stage Weight Due to Engine Weight

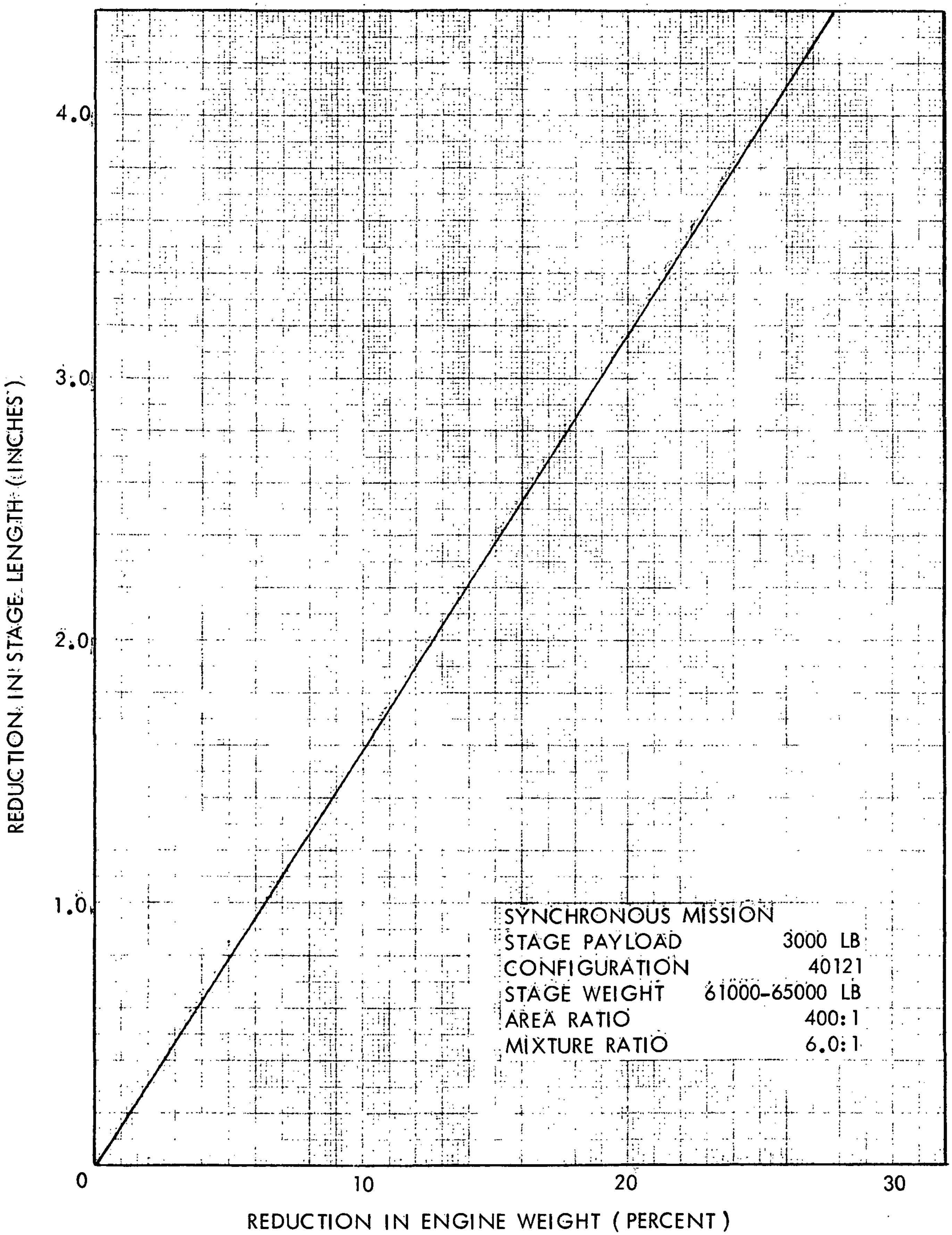

Figure 3-12. Variation of Stage Length Due to Engine Weight

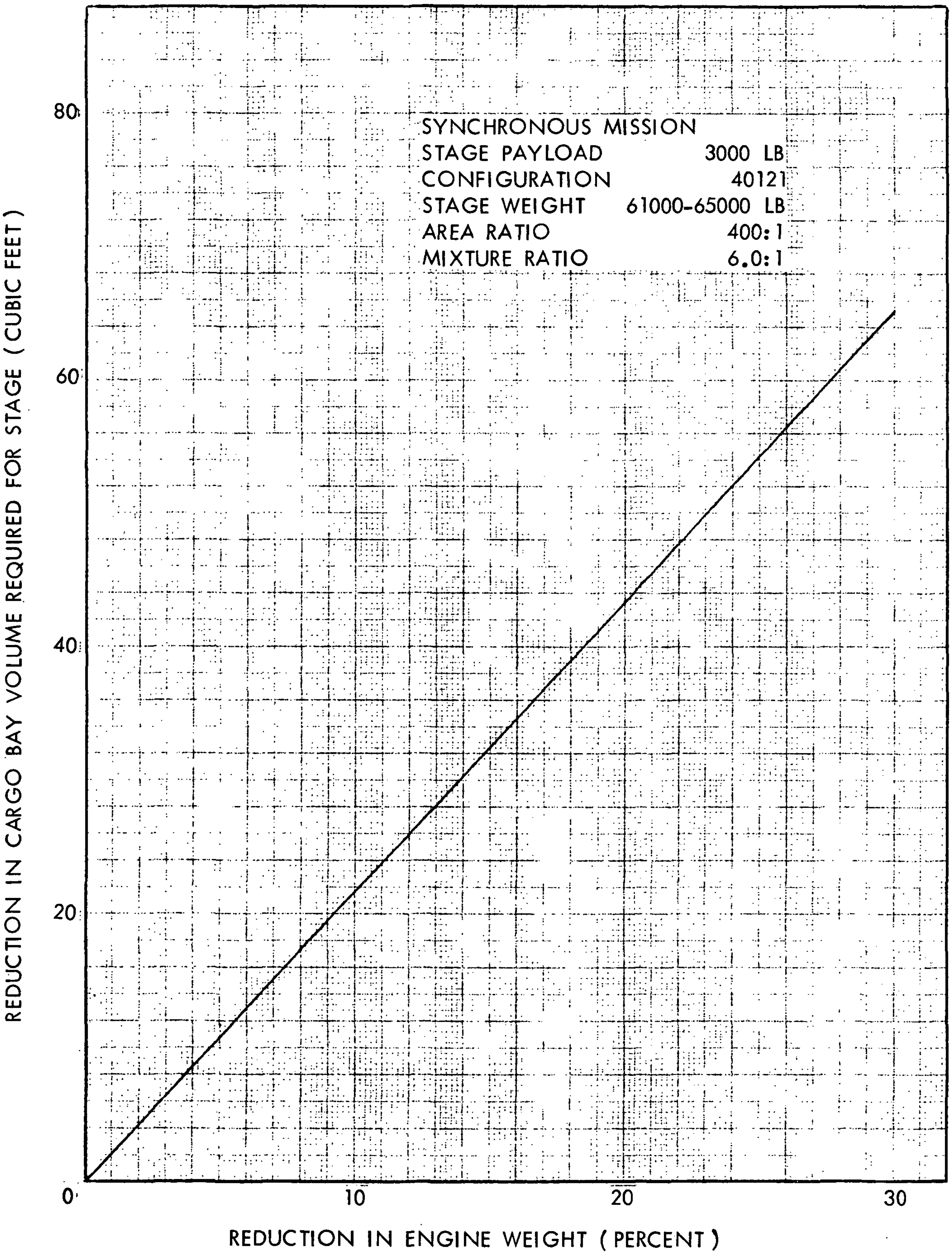

Figure 3-13. Variation of Stage Volume Due to Engine Weight

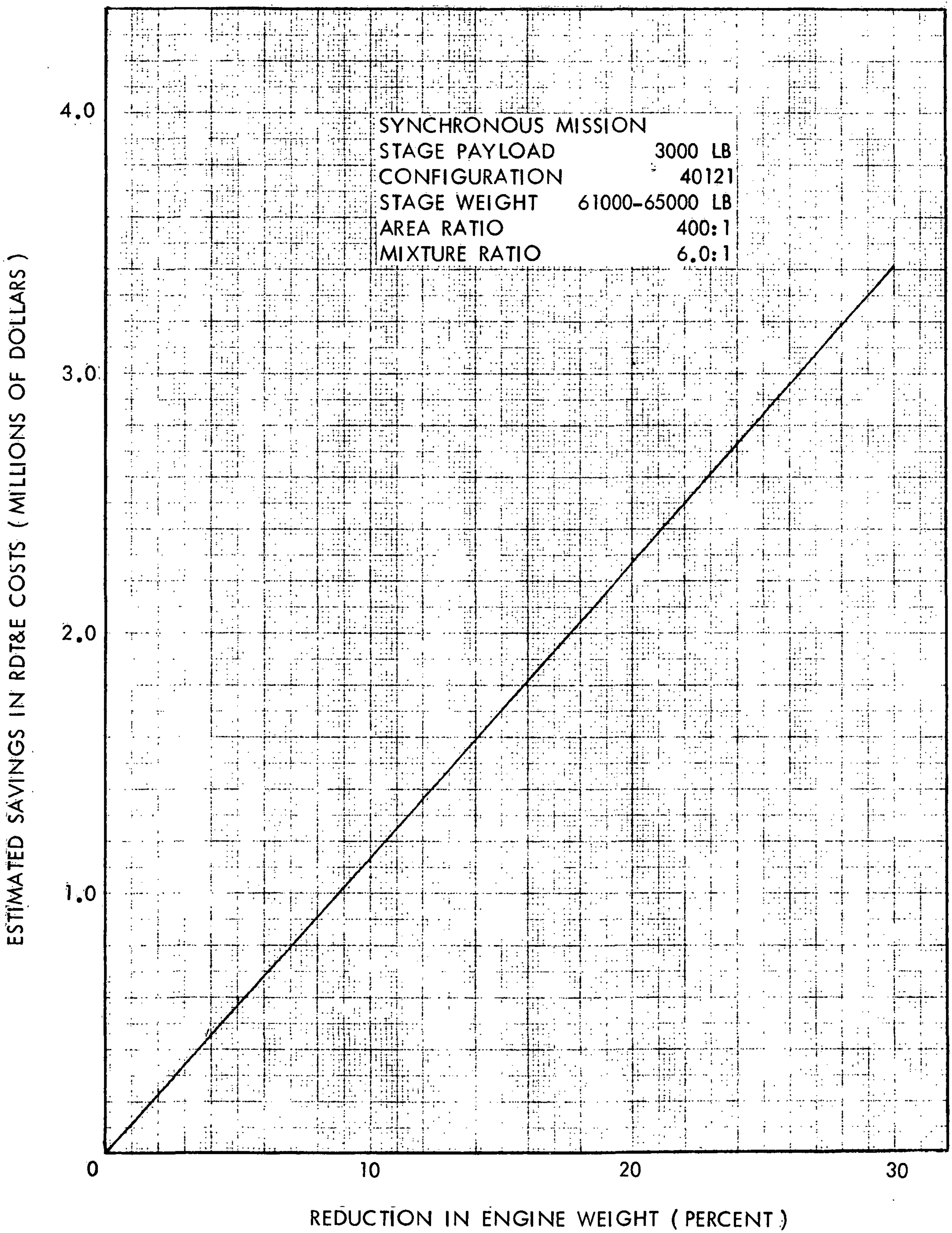

Figure 3-14. Variation of RDT&E Cost Due to Engine Weight

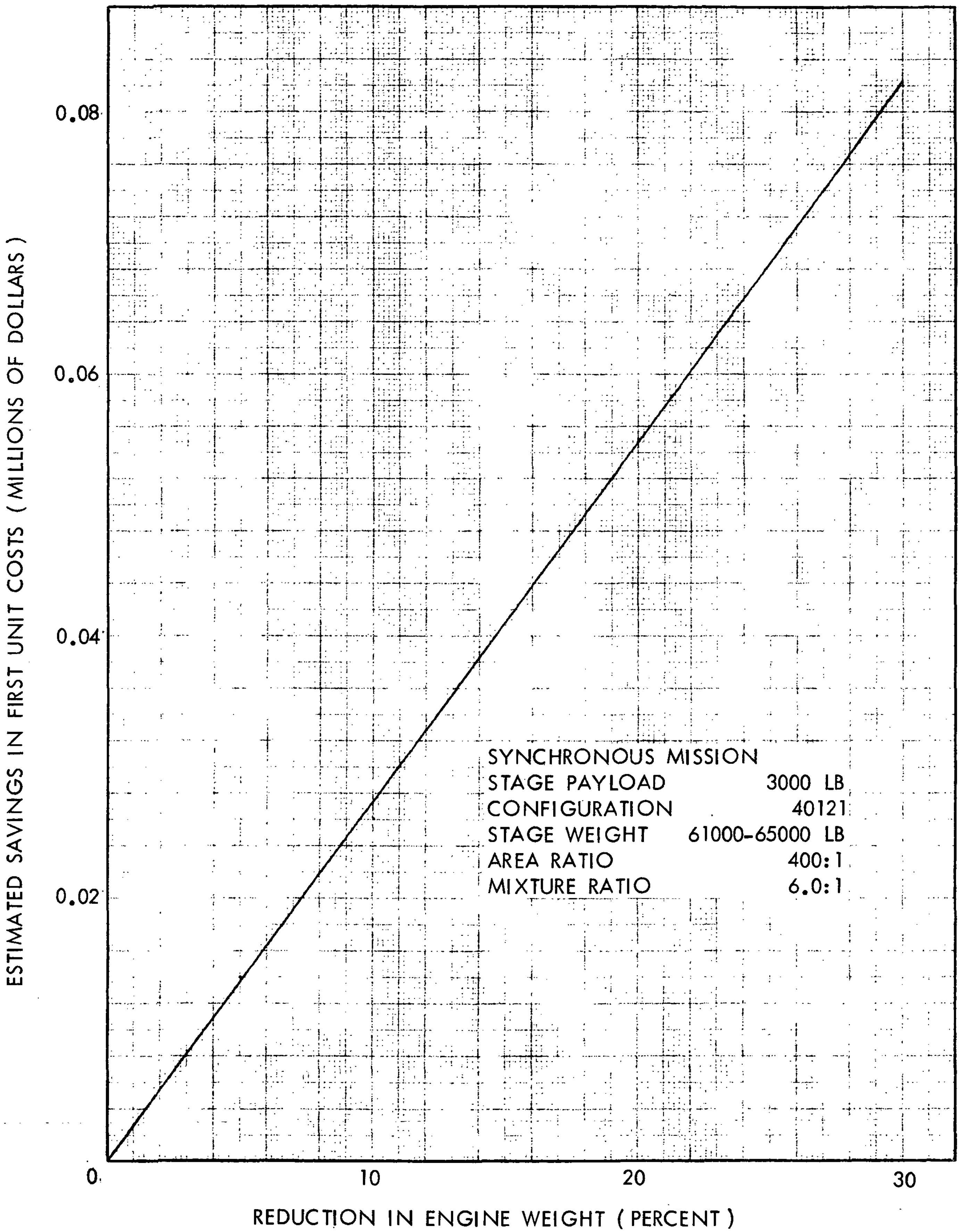

Figure 3-15. Variation of First Unit Cost Due to Engine Weight

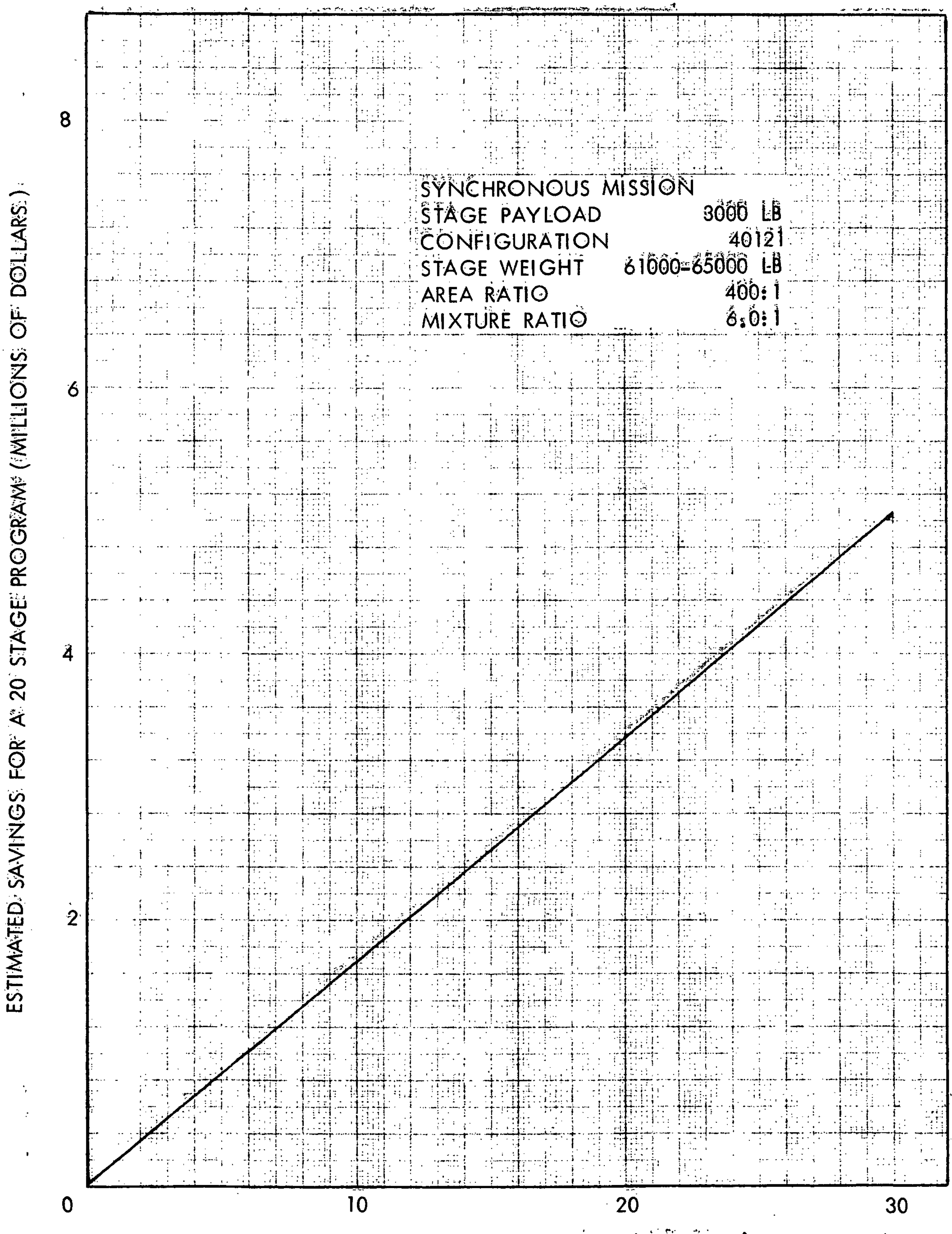

Figure 3-16. Variation of Program Cost Due to Engine Weight

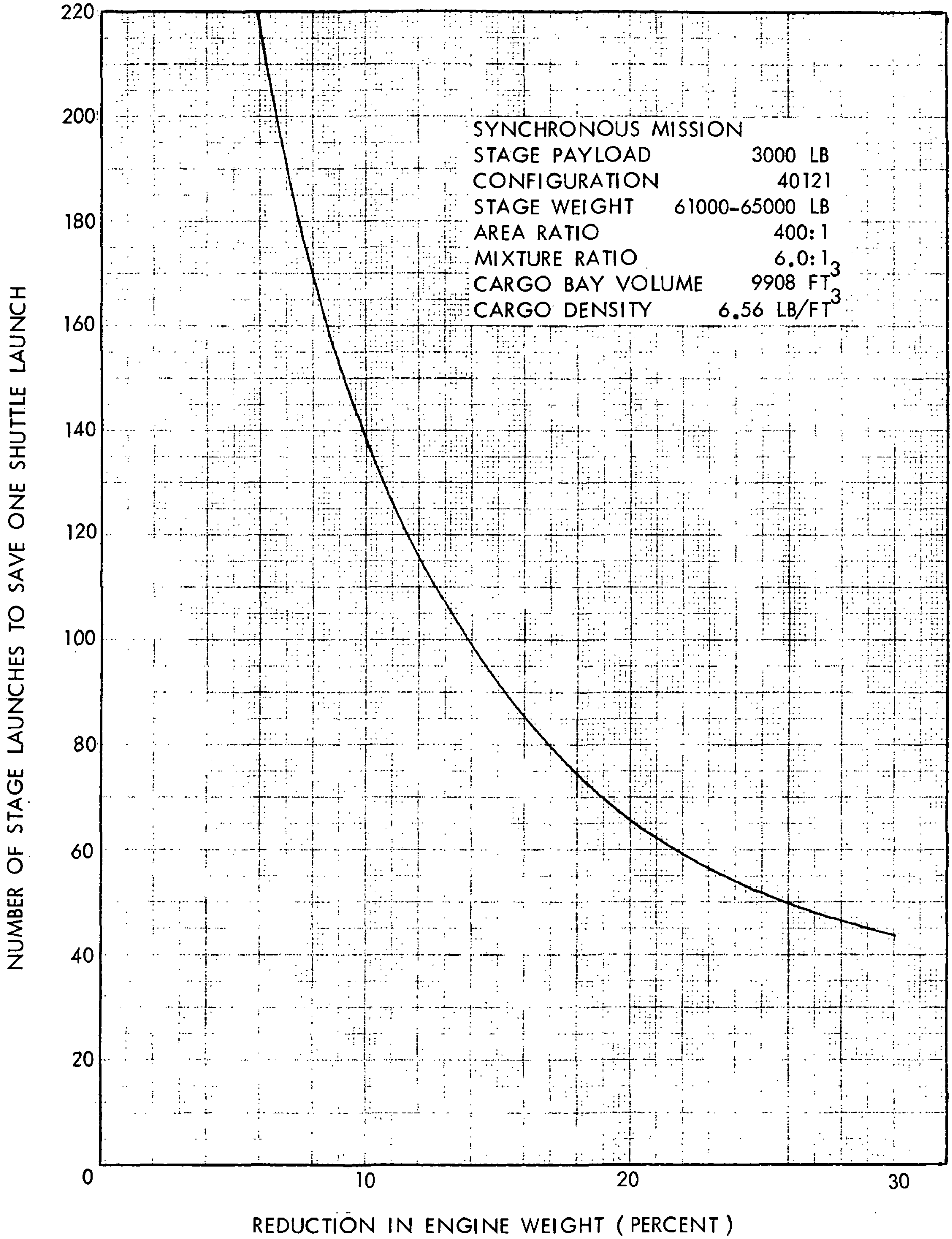

Figure 3-17. The Number of Stage Launches Required to Save One Shuttle Flight (Engine Weight)

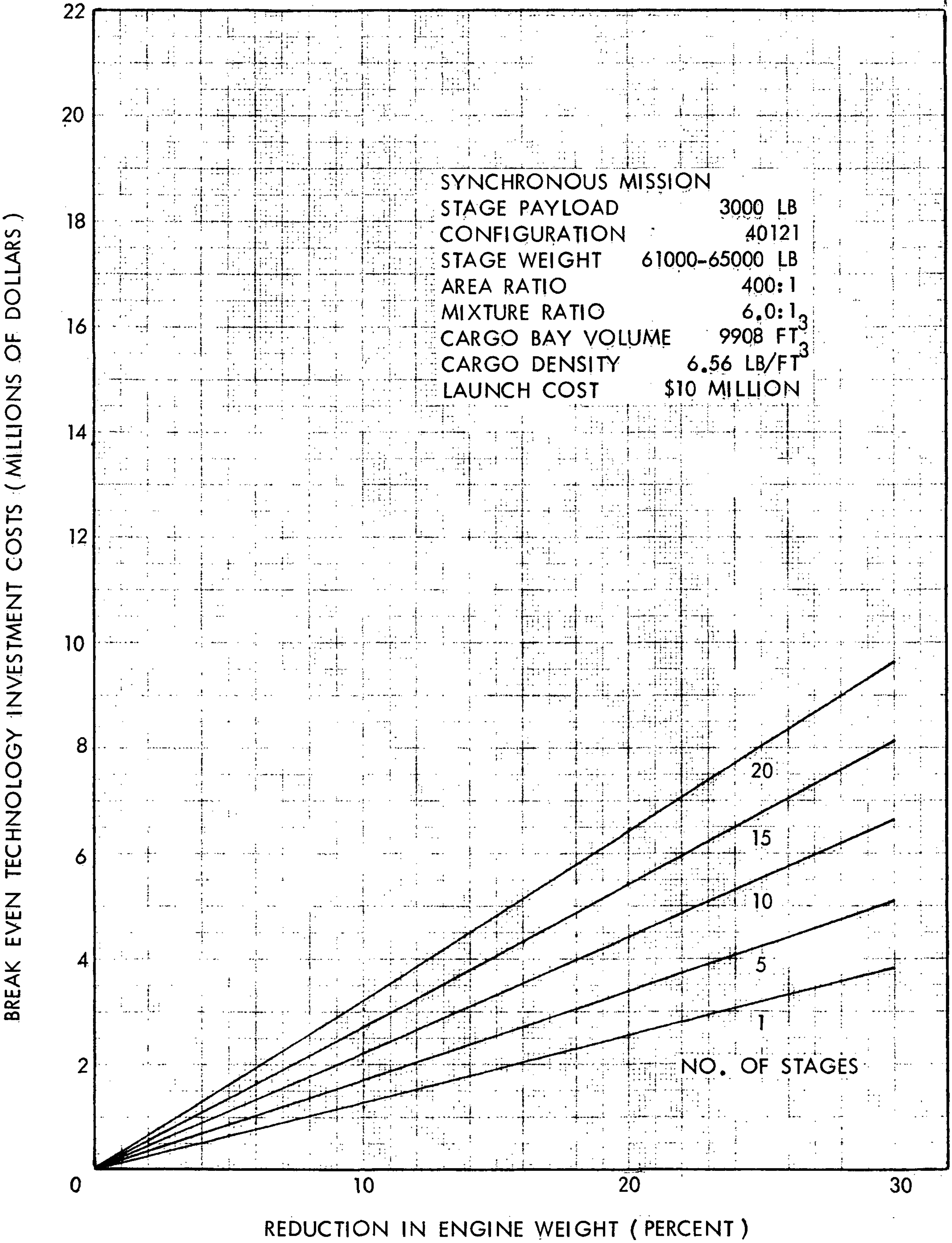

Figure 3-18. Break Even Costs for Engine Weight

but with a higher chamber pressure (3000 psi) and a lower area ratio (200:1), was selected for the new baseline for this analysis. Table 3-11 presents the more pertinent data for this reference stage.

The results of the area ratio analysis are presented in figures 3-19 through 3-26. The technology break-even costs shown in figure 3-26 are only slightly less than those obtained for increases in chamber pressure. The reason for this is the lower cost savings obtained and not the fact that the stages become longer as the engine nozzle expansion ratio is increased, as shown in figure 3-20. This increase in stage length is due entirely to the longer engines resulting from the higher area ratios, and not the height of the stage (the distance from the gimbal point to the stage-payload interface), which decreases. As discussed in subsection 3.2, the change in stage length becomes important only when small stages are being considered for use with the shuttle.

3.6 MISCELLANEOUS SUBSYSTEM WEIGHT

An analysis was undertaken to determine the influence of miscellaneous subsystem weights (that is: fixed inert weights, such as astrionics, etc.) on stage size and cost. The results of this analysis are presented in figures 3-27 through 3-34.

These data were based on equal percent reductions in all the miscellaneous subsystem weights. Because of the limited duration of this study, no attempt was made to determine the driving subsystem, or the effects of individual subsystems.

The variations in stage size are illustrated in figures 3-27, 3-28 and 3-29, which show the effect of subsystem weight on stage weight, length and volume, respectively. As would be expected because of the large amount of fixed inert weight (1900 pounds) assumed, these figures indicate that stage weight is quite sensitive to subsystem weight.

The effect of decreased subsystem weight on RDT&E, TFU and program costs is depicted in figures 3-30, 3-31 and 3-32, respectively. These data indicate that substantial cost savings might be realized through the reduction of subsystem weight. However, because the cost savings presented in these figures do not include the cost required to develop the lighter weight technology (RDT&E), nor the higher investment cost associated with lighter systems (such as using integrated circuits instead of transistors in the electronic systems), the actual savings will be less than indicated.

However, the data do give a true indication of the savings which might be obtained through the elimination of components, such as redundant valves, a secondary transponder, etc.

The number of stage launches required to save a single shuttle flight and the break-even technology investment costs (technology costs = total savings), are presented in figures 3-33 and 3-34, respectively. Figure 3-34 shows that substantial investments can be made in order to reduce the subsystem weights a slight amount, and still break even.

Table 3-11. Baseline Stage Data for Area Ratio Analysis

PAYLOAD (UP/DOWN)	3K / 3K
TOTAL STAGE WEIGHT	64254
INERT STAGE WEIGHT	(5454)
STRUCTURE, TANKS, ETC.	2304
METEOROID PROTECTION	0
THERMAL PROTECTION	188
PROPULSION	566
MISCELLANEOUS SUBSYSTEMS	1900
CONTINGENCY	496
FLUIDS INVENTORY	(58800)
IMPULSE PROPELLANT	56892
NON-IMPULSE PROPELLANT	328
VENTED	0
RESIDUAL	1580
ENGINE CHARACTERISTICS	
THRUST	16814
SPECIFIC IMPULSE	467.6
AREA RATIO	200:1
CHAMBER PRESSURE	3000
MIXTURE RATIO	6.0:1
COST DATA* ($M)	
RDT&E	683.861
FIRST UNIT	17.659
PROGRAM (20 STAGES)	1032.894

*Does not include technology and normal operations cost, but includes propellant costs.

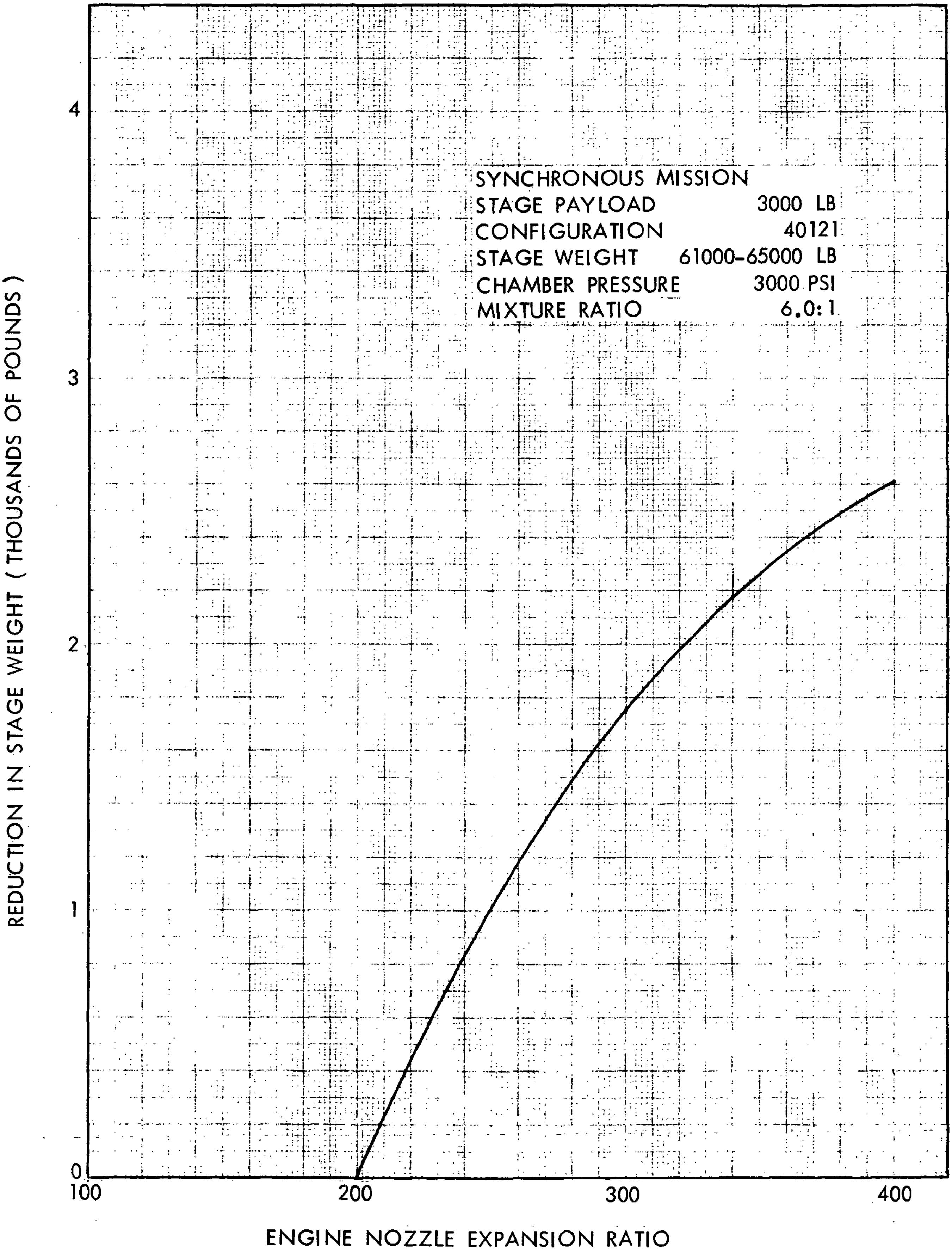

Figure 3-19. Variation of Stage Weight Due to Area Ratio

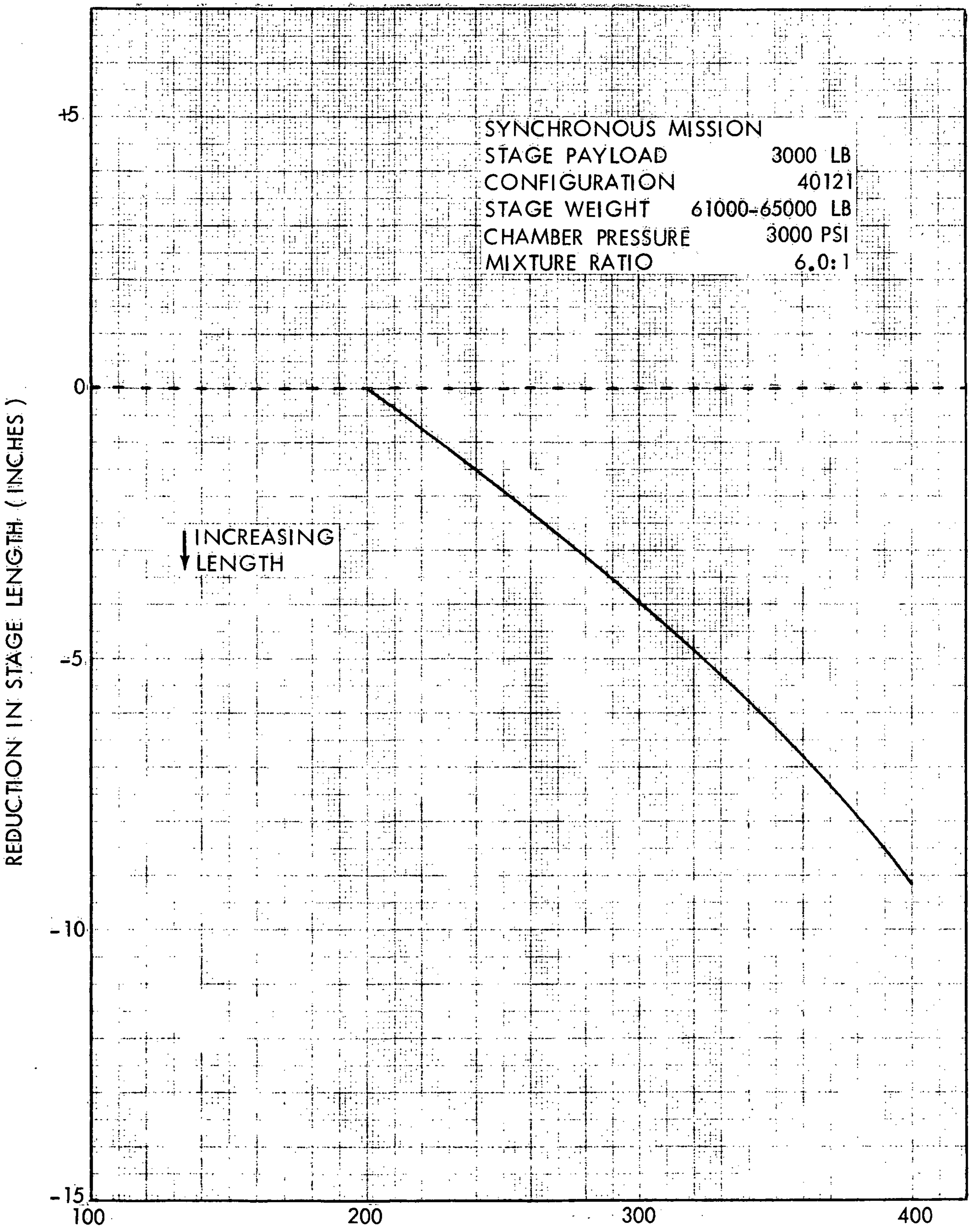

Figure 3-20. Variation of Stage Length Due to Area Ratio

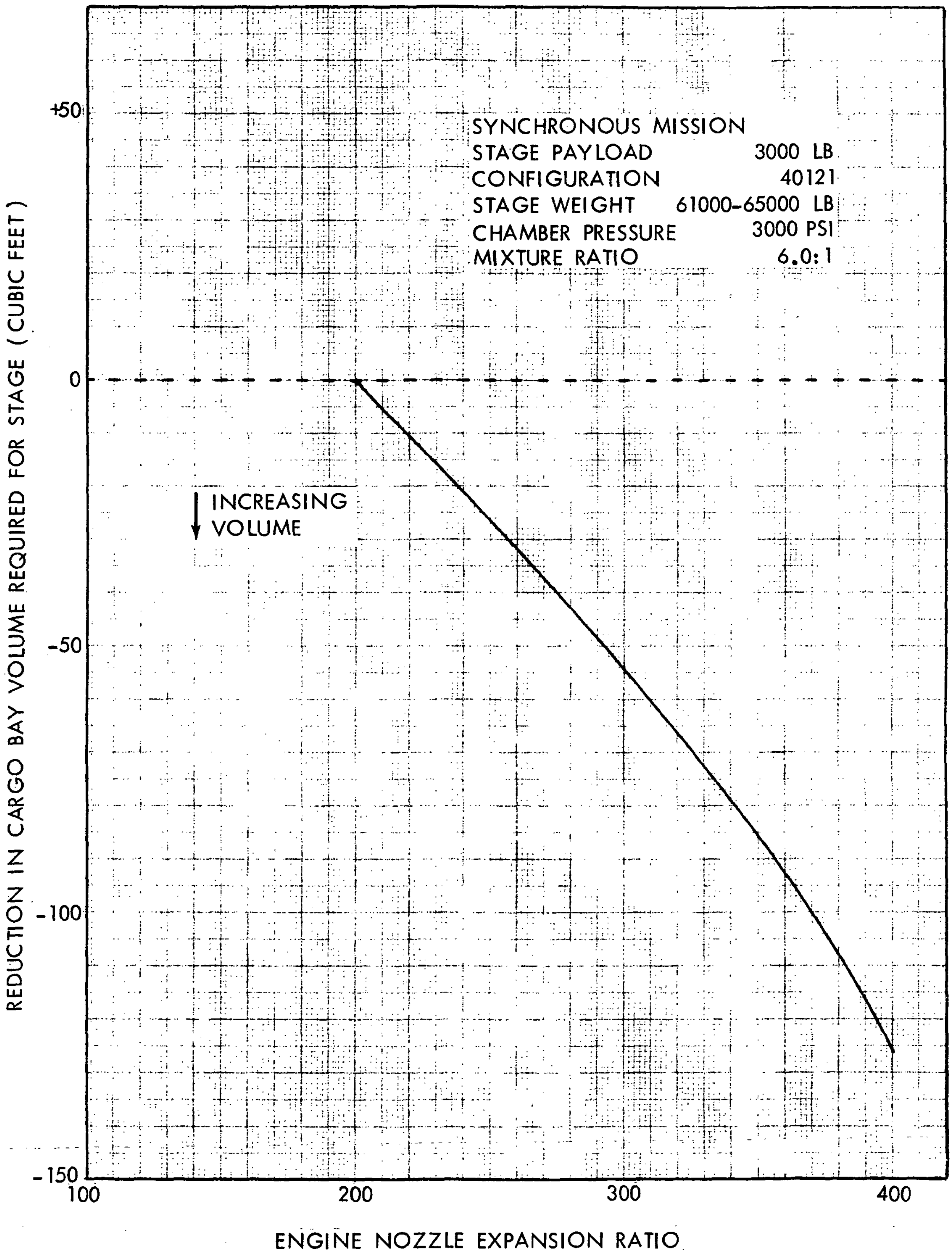

Figure 3-21. Variation of Stage Volume Due to Area Ratio

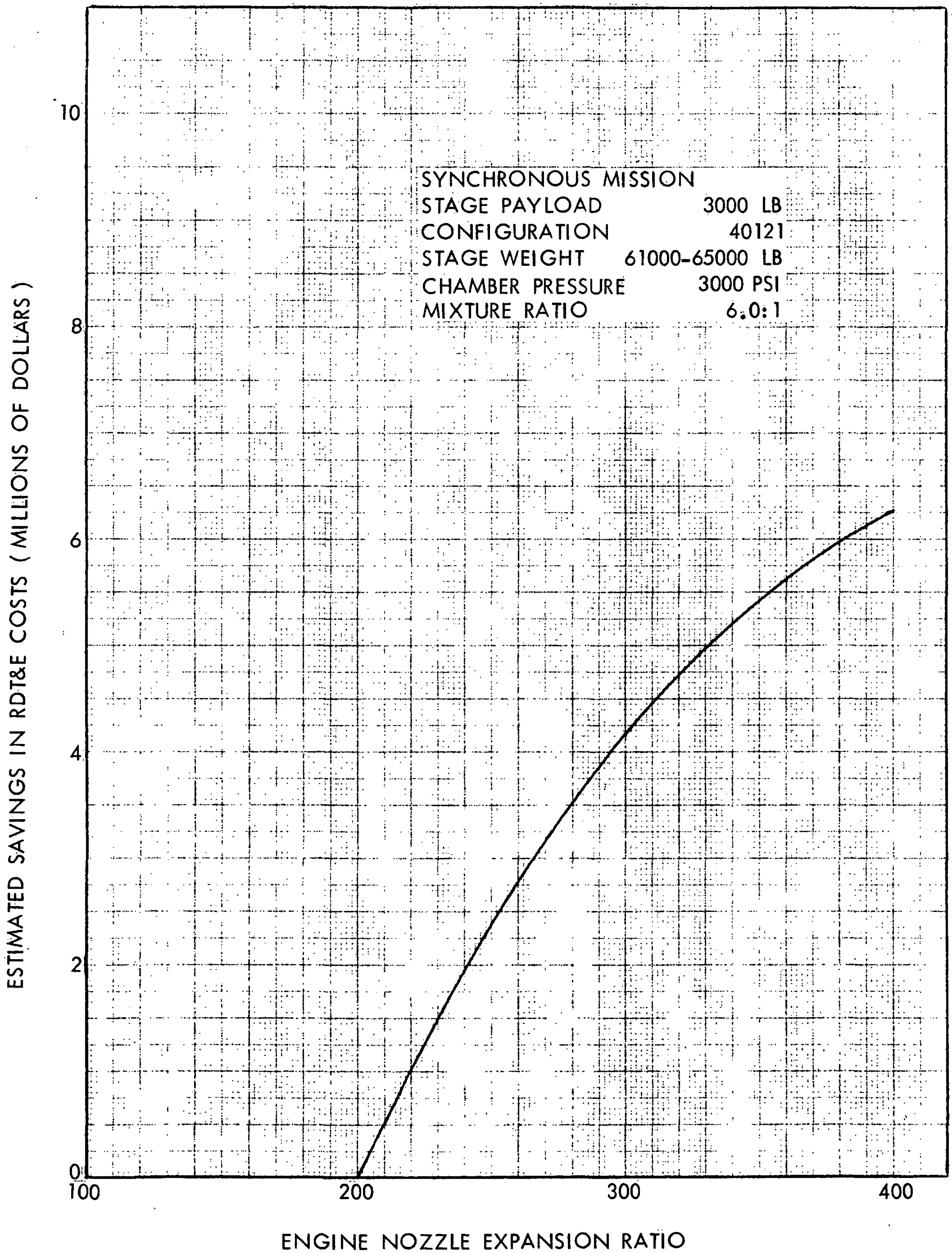

Figure 3-22. Variation of RDT&E Cost Due to Area Ratio

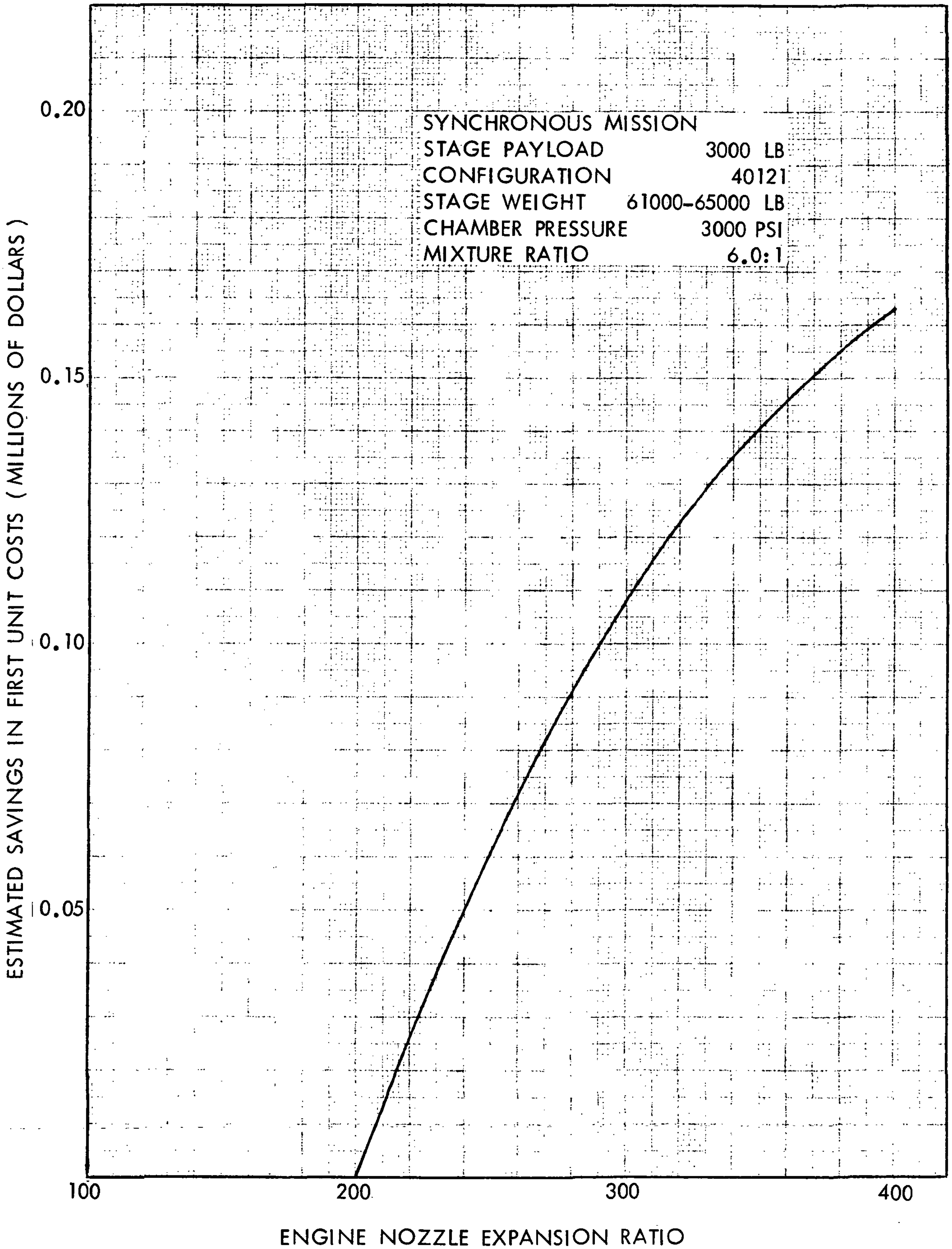

Figure 3-23. Variation of First-Unit Cost Due to Area Ratio

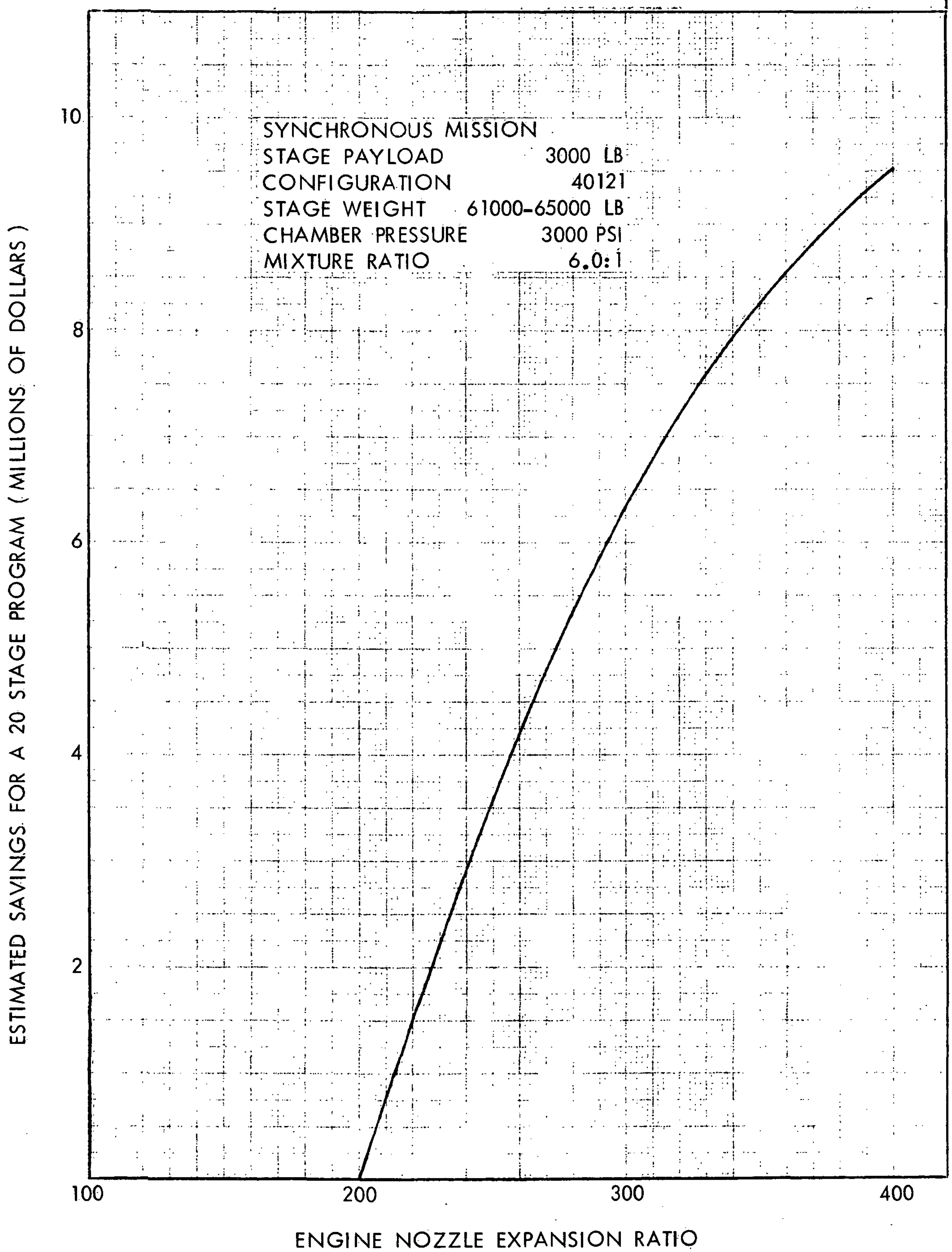

Figure 3-24. Variations of Program Cost Due to Area Ratio

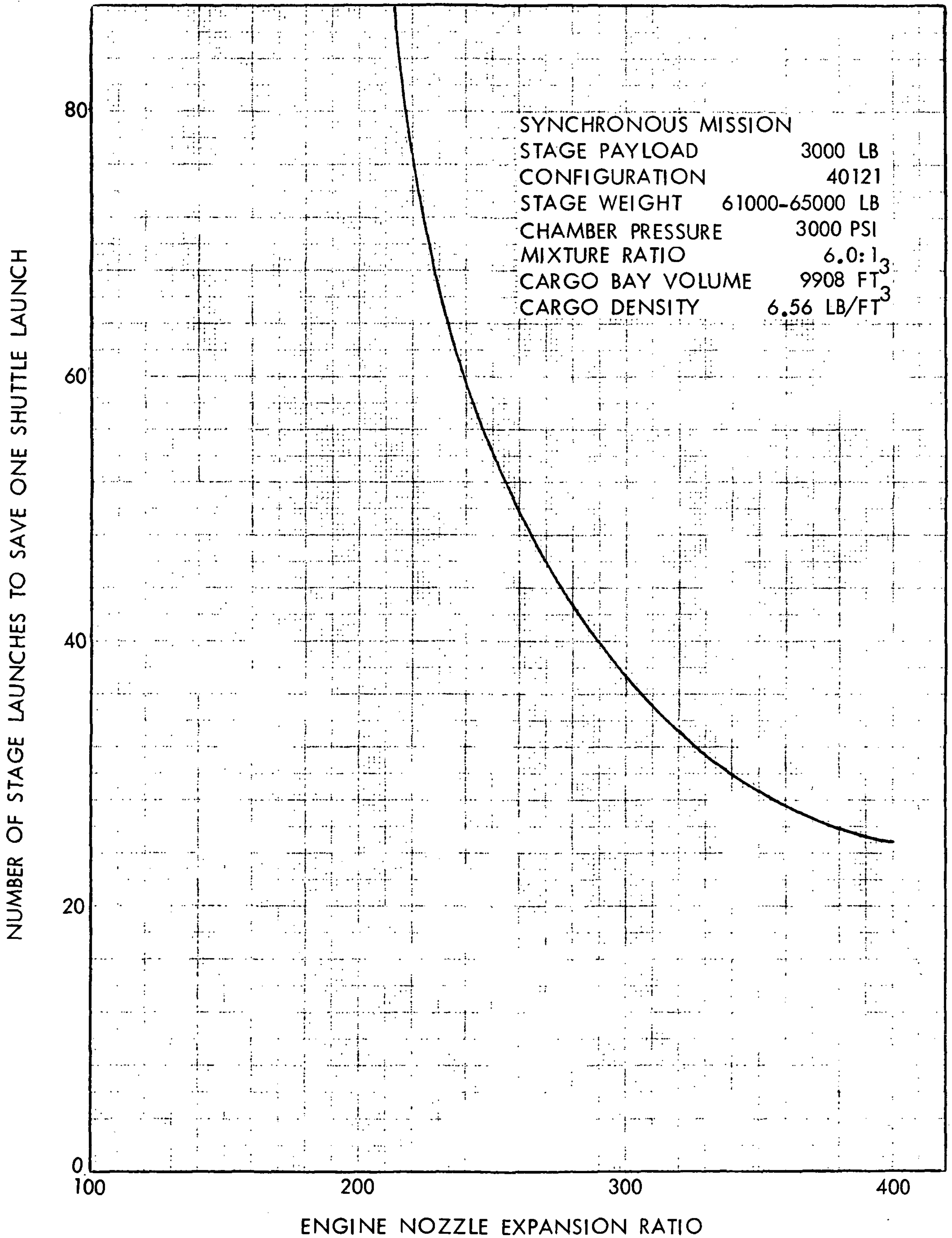

Figure 3-25. The Number of Stage Launches Required to Save One Shuttle Flight (Area Ratio)

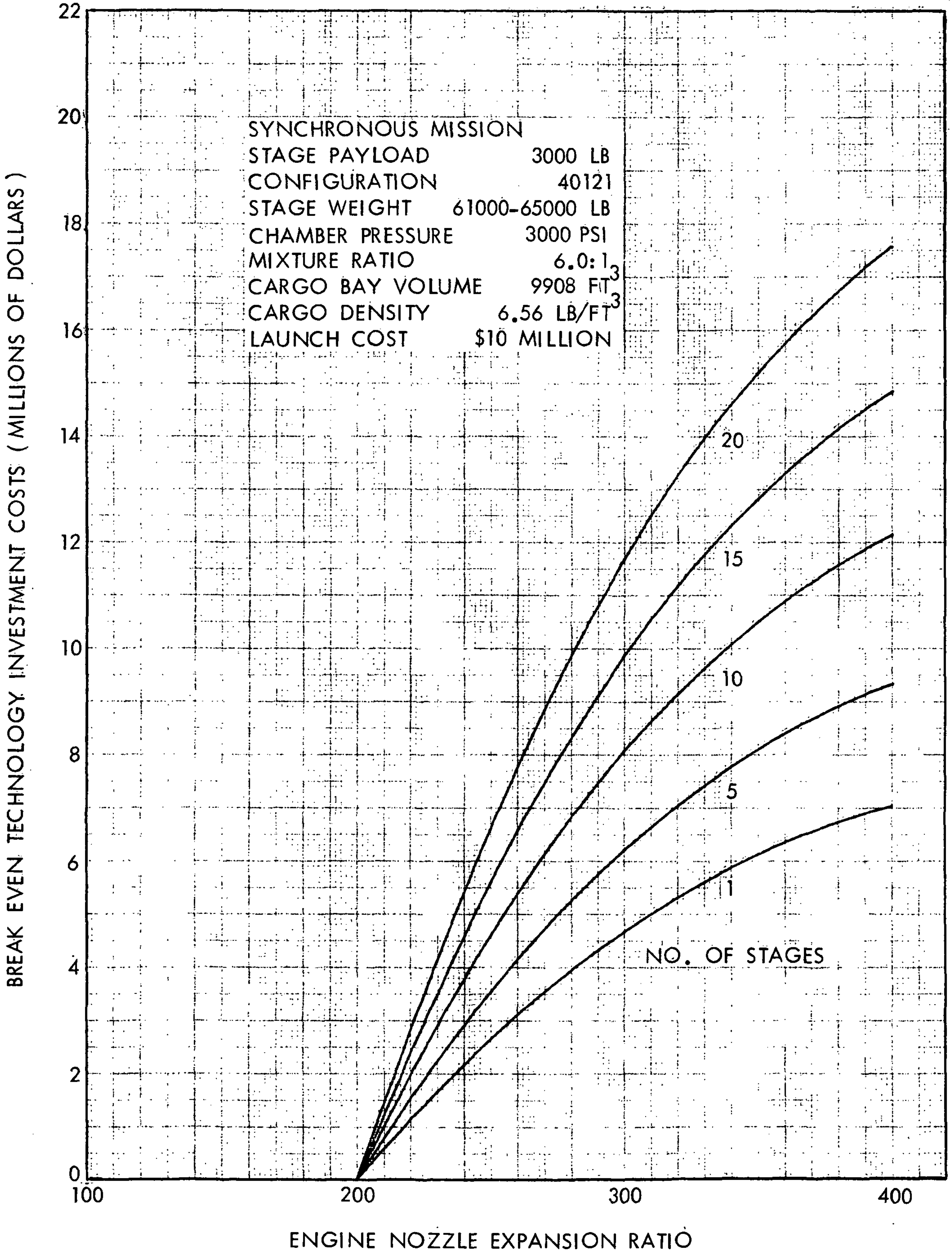

Figure 3-26. Break-Even Technology Costs for Area Ratio

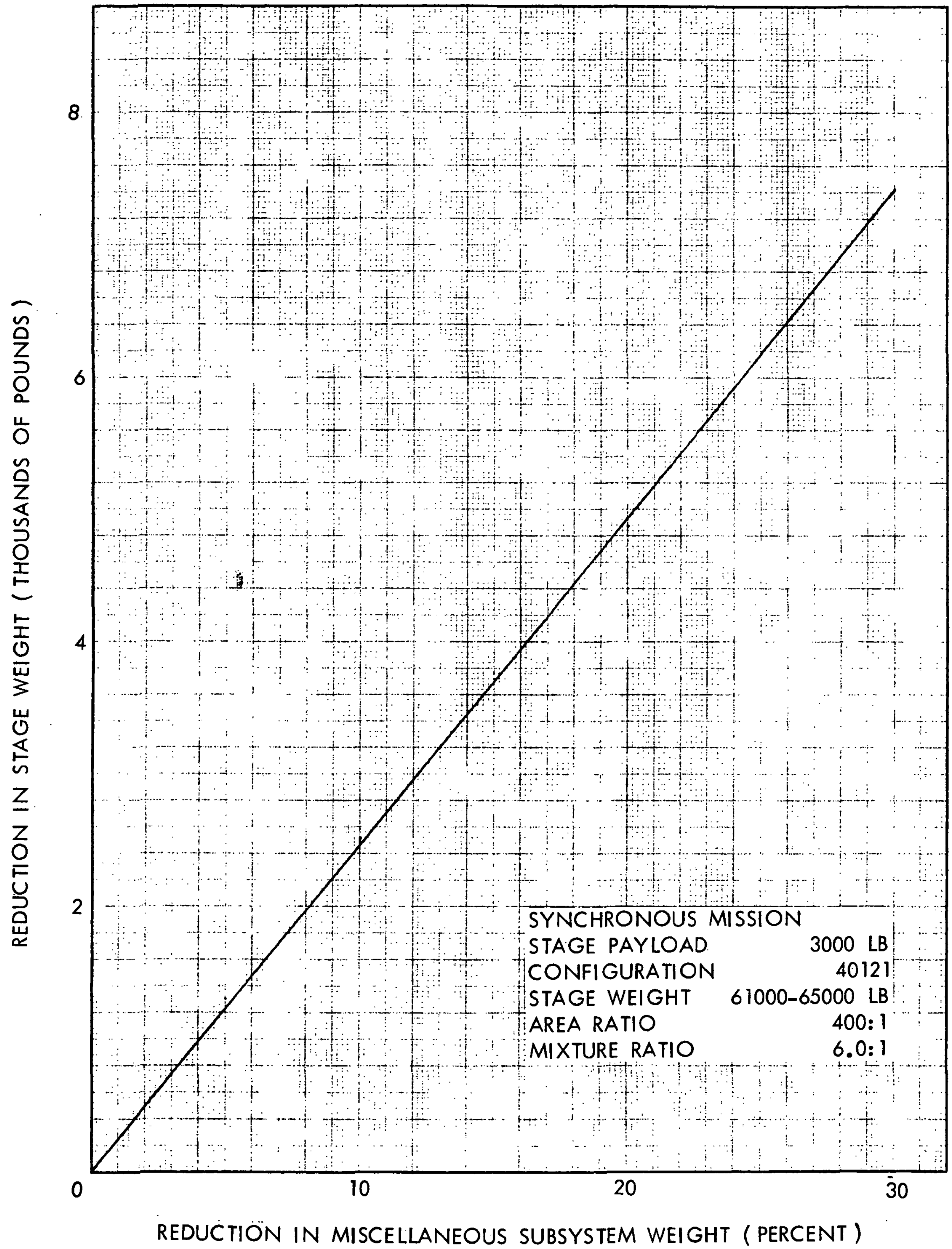

Figure 3-27. Variation of Stage Weight Due to Subsystem Weight

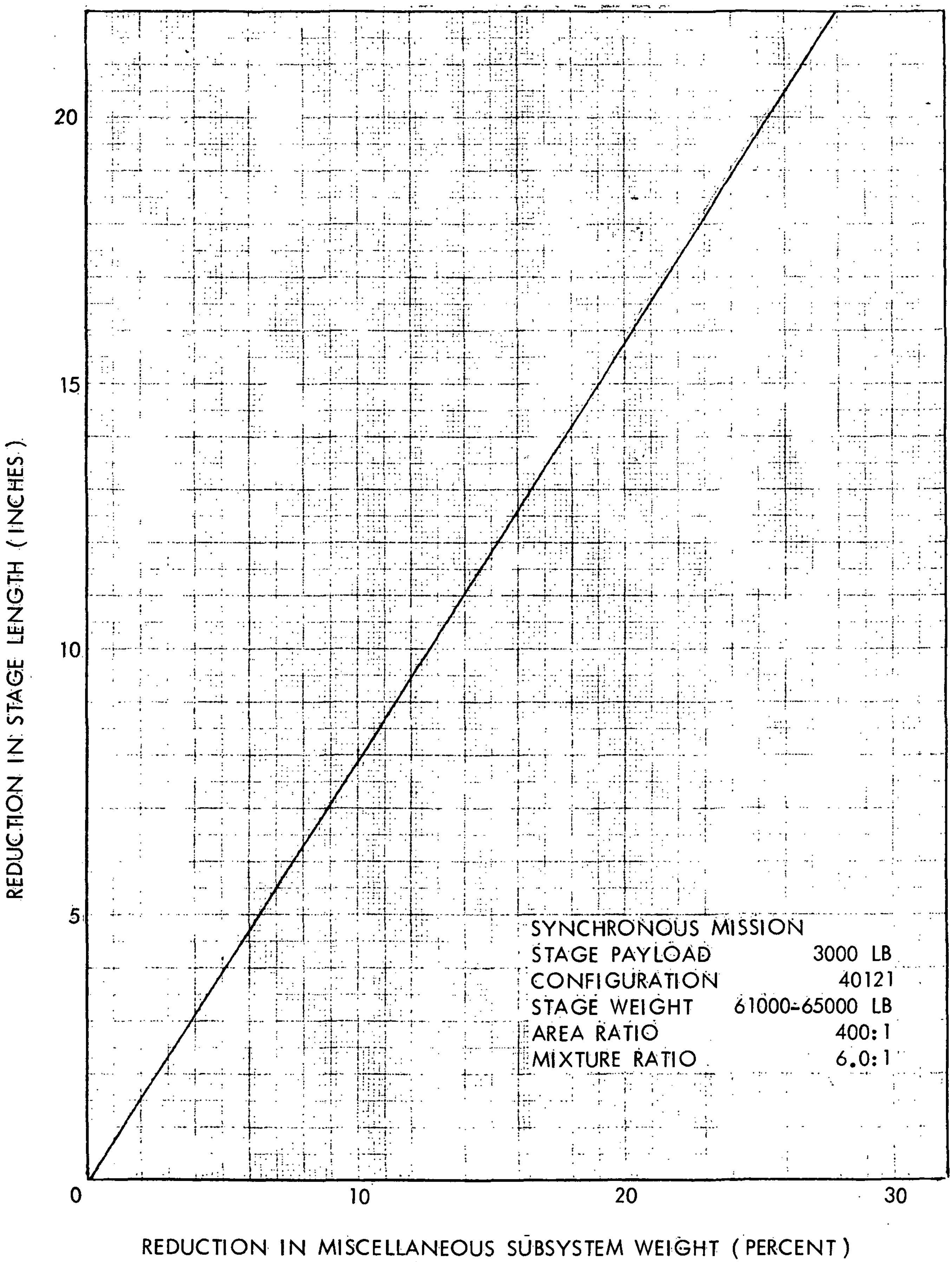

Figure 3-28. Variations of Stage Length Due to Subsystem Weight

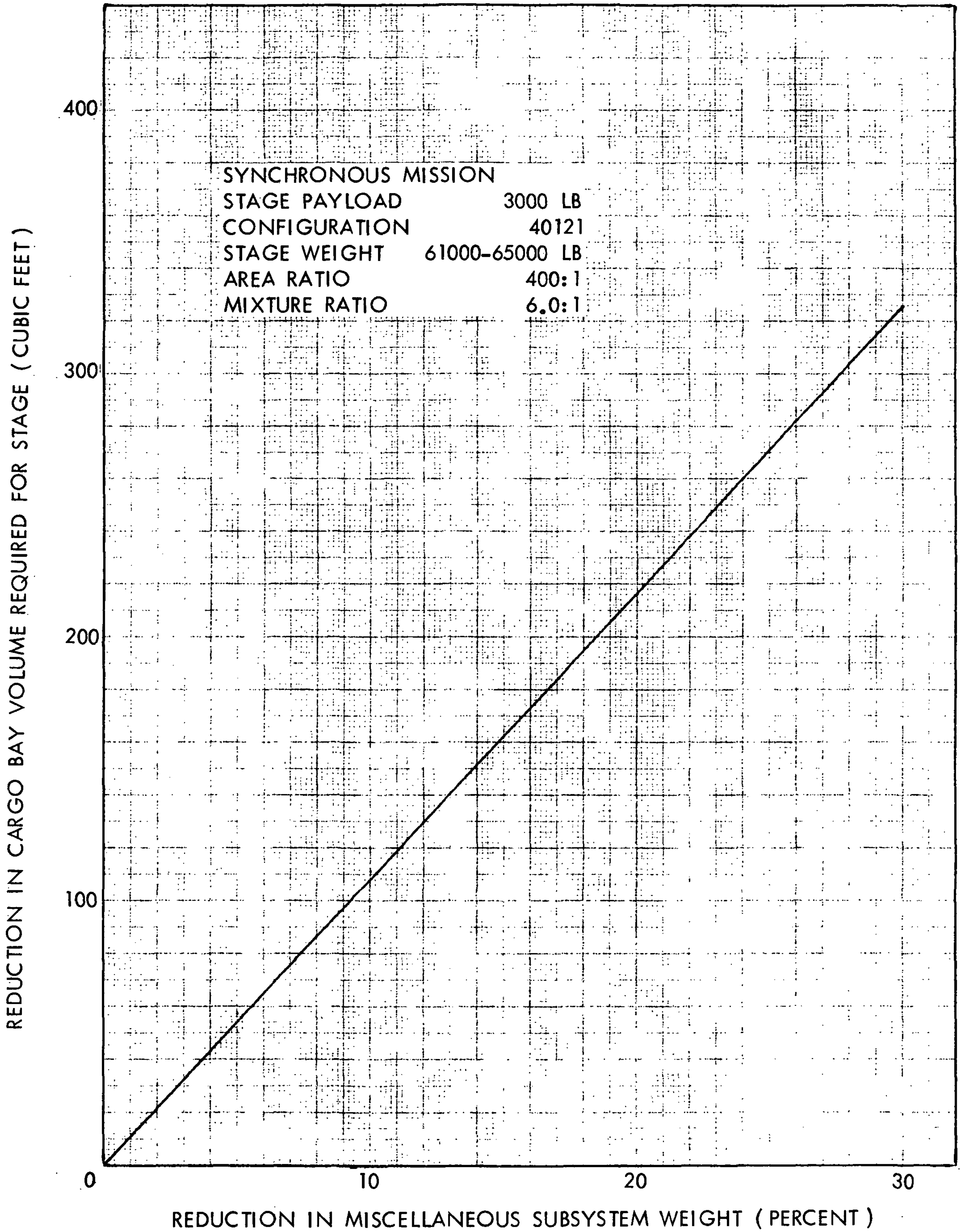

Figure 3-29. Variations of Stage Volume Due to Subsystem Weight

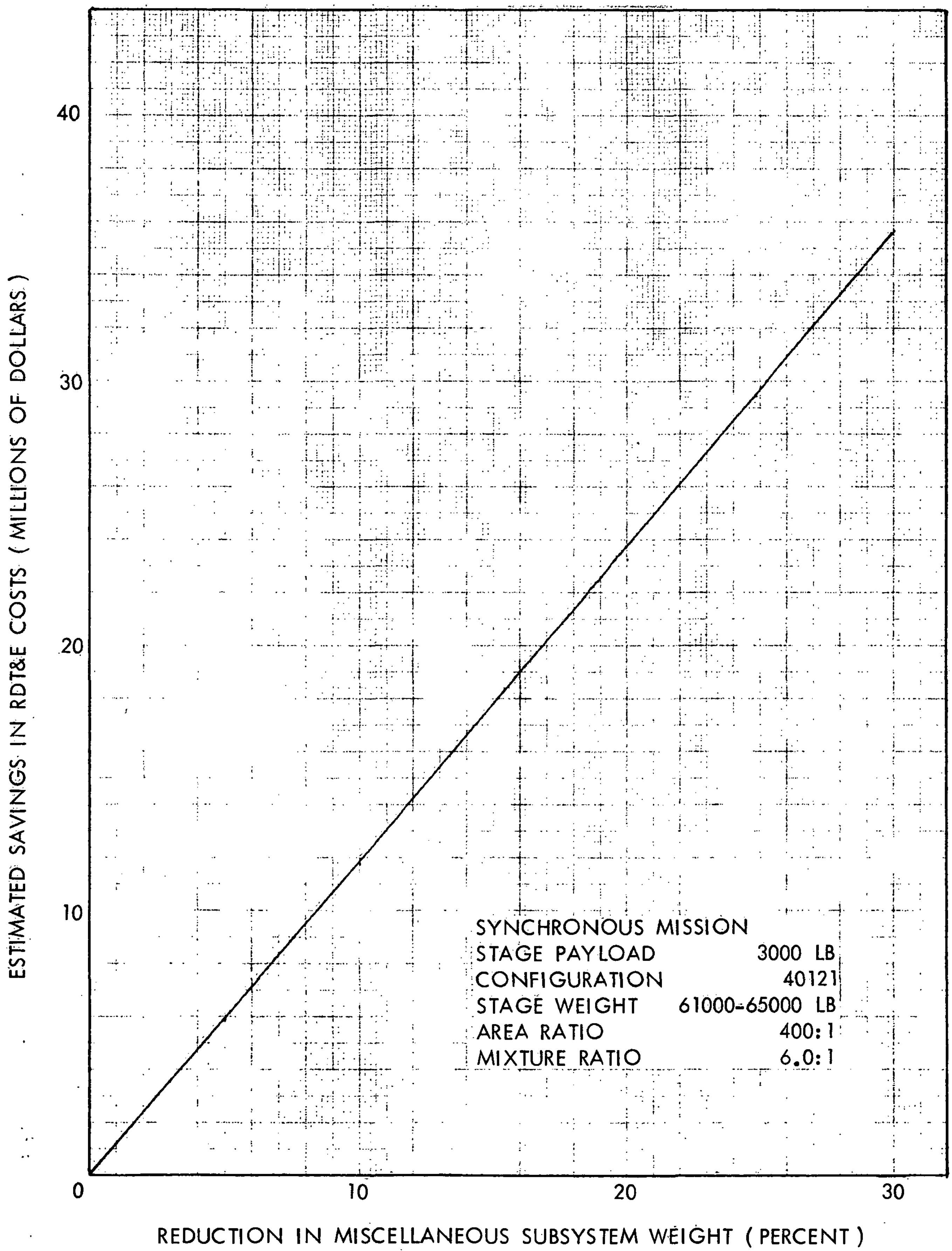

Figure 3-30. Variations of RDT&E Cost Due to Subsystem Weight

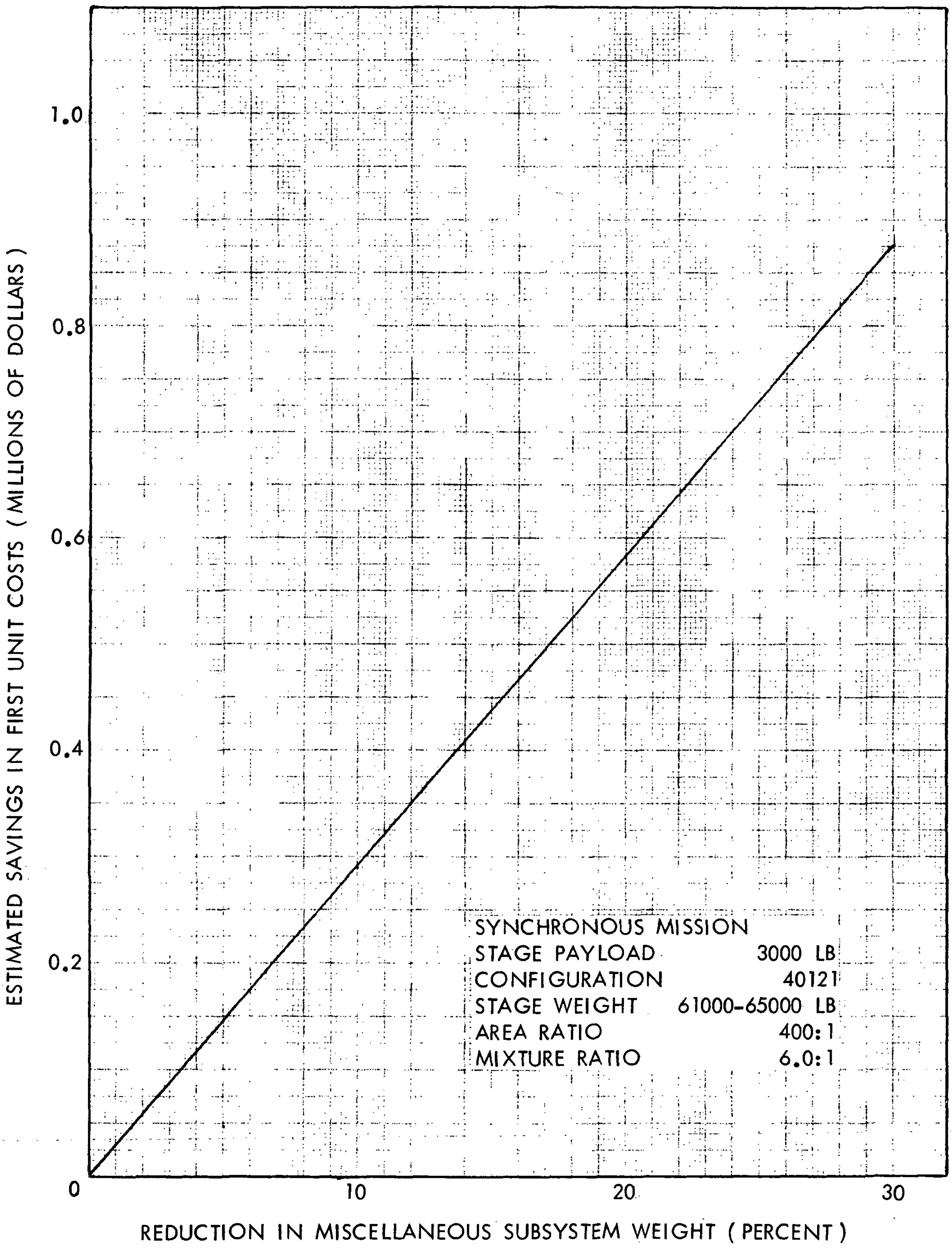

Figure 3-31. Variations of First-Unit Cost Due to Subsystem Weight

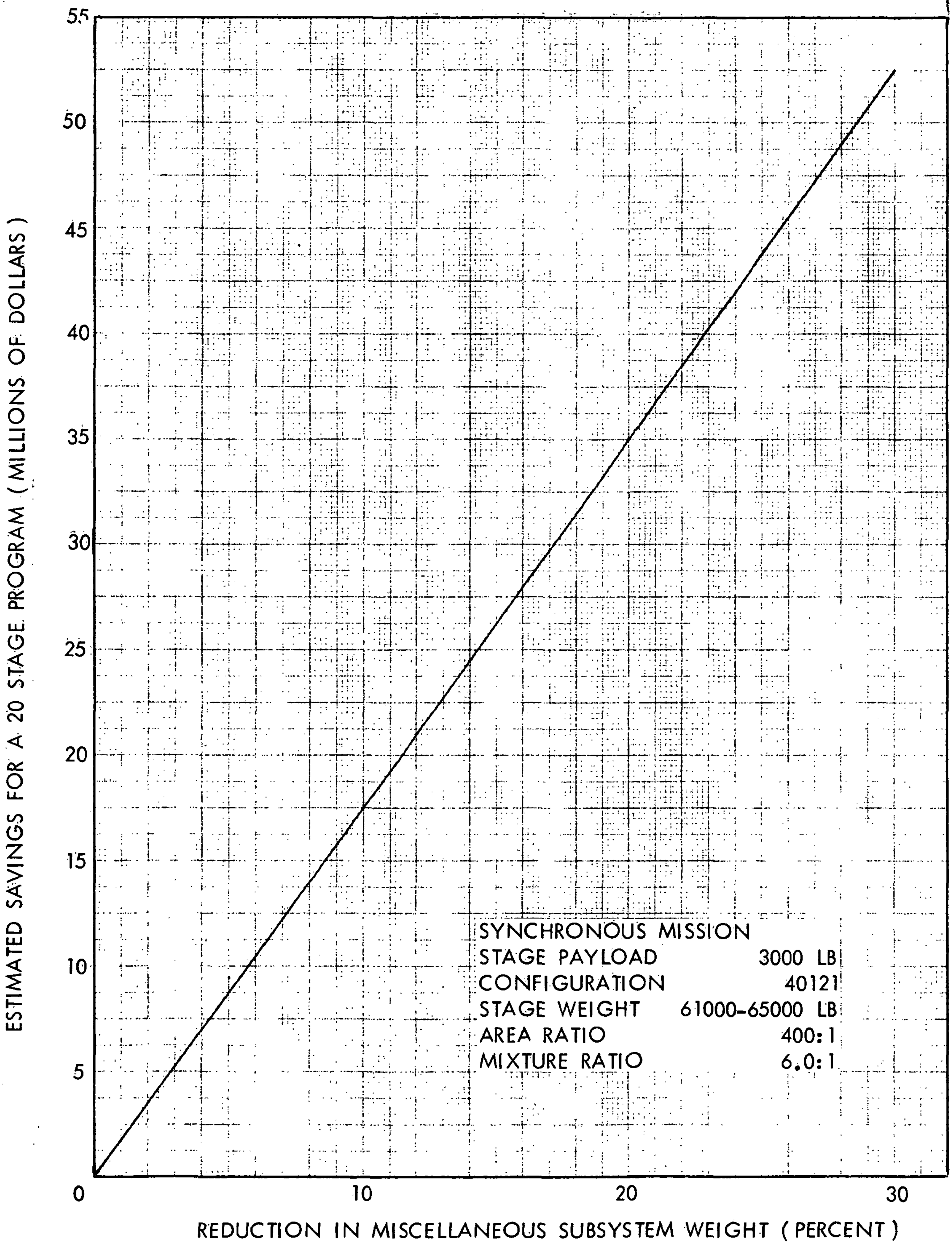

Figure 3-32. Variations of Program Cost Due to Subsystem Weight

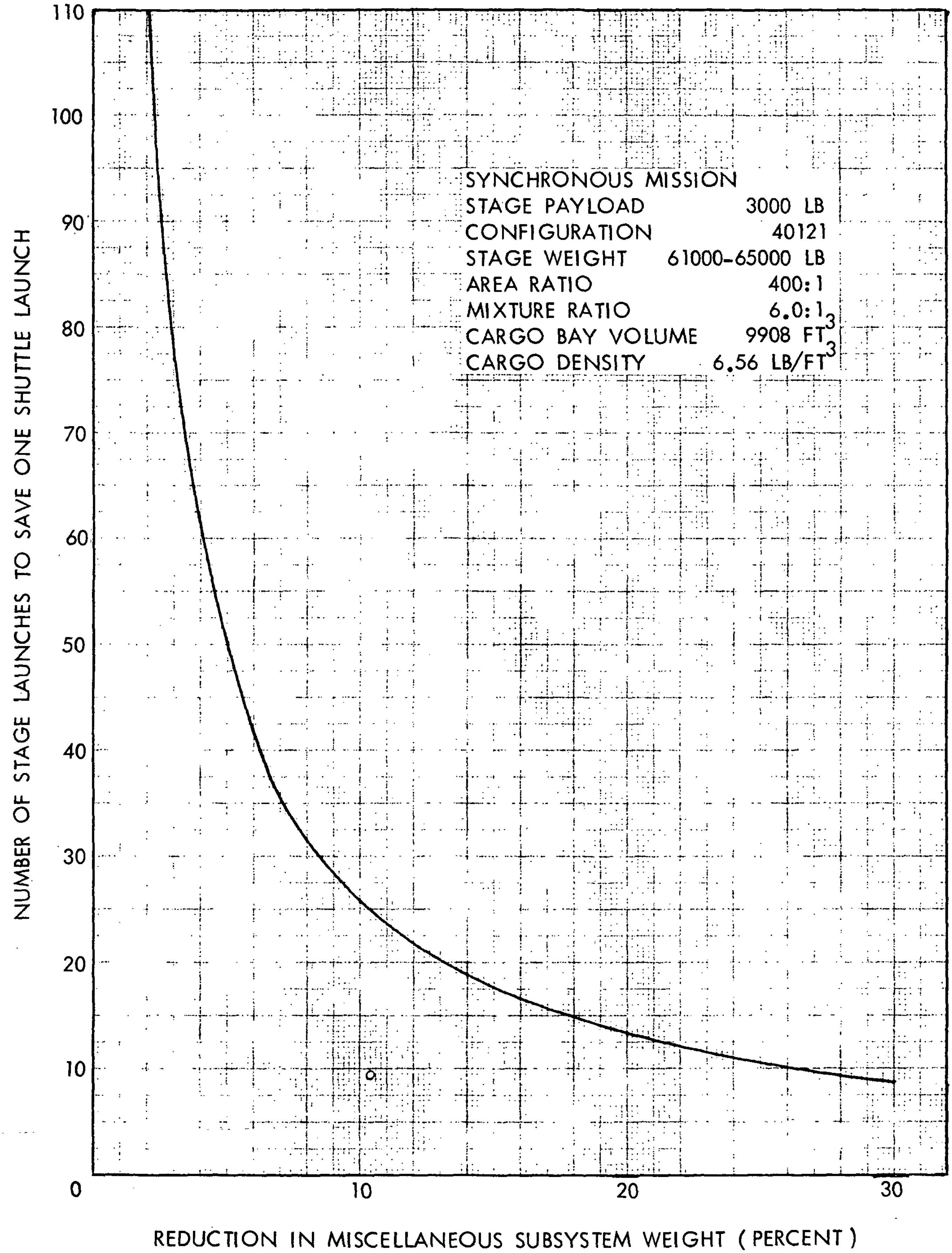

Figure 3-33. The Number of Stage Launches Required to Save One Shuttle Flight (Subsystem Weight)

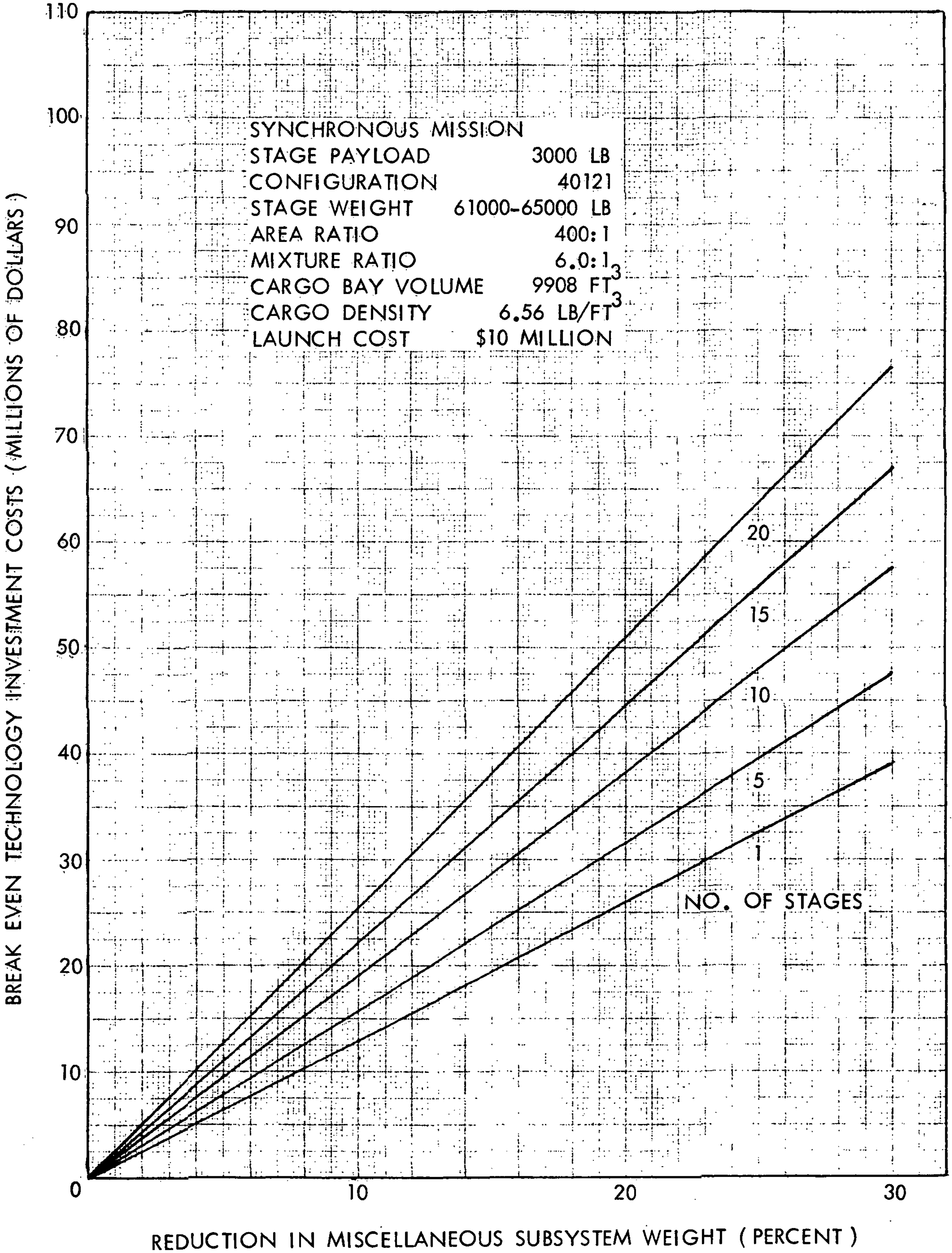

Figure 3-34. Break-Even Technology Costs for Subsystem Weight

3.7 TANK INSULATION THERMAL CONDUCTIVITY

The influence of the propellant tank insulation thermal conductivity on stage size and cost was also analyzed, and the corresponding break-even technology costs determined. The results which are presented in figures 3-35 through 3-42 show that both improved thermal insulation and reduced engine weight have about the same effect on stage size and cost. Although the weight of the insulation is about half that of the engine, slightly greater savings are obtained through improved insulation properties. The reason for this is that the thermal conductivity not only influences the thickness of insulation required, but also the propellant tank design pressure, and to a certain extent, the tank's ullage volume. These, together with the propellant load, determine the tank size and weight. In addition, the thermal conductivity of insulation can also have an effect on the pressurant requirements, which determine the weight of the pressurization system. Hence, the thermal conductivity of the tank insulation affects other systems whose total weights exceed that of the engine.

3.8 PRIME STRUCTURE WEIGHT

The last analysis conducted as part of this task assignment was the determination of the effect of new materials, e.g., the use of honeycomb or composite instead of aluminum sheet stringer, for the prime structure components. The only portions of the structure considered in this analysis were the load carry shell (which surrounds the single hydrogen tank) and the thrust cone. Other areas, namely the tank supports, propellant feedlines, etc., were not considered. The influence of prime structure weight on stage size and cost was established by sizing stages with prime structures which weighed various percents of the baseline aluminum sheet/stringer weight. Costs were determined for the resized stages, taking into account the costs of the respective technologies. An analysis was then undertaken to determine the structural weight reduction which might be obtained through the use of aluminum honeycomb and an advanced composite material. The results of these analyses are presented in figures 3-43 through 3-49.

The effect of prime structure on stage size is presented in figures 3-43, 3-44, and 3-45, which show the variation of stage weight, length and volume, respectively. The range of weight reduction which might be expected, is indicated in each of these figures for both the honeycomb and the composite. Figure 3-43 shows that structural weight reductions of 10 to 15 percent and 25 to 35 percent can be expected over the aluminum sheet/stringer for honeycomb and composites, respectively. These correspond to decreases in stage weights of 800 to 1200 pounds and 2100 to 2800 pounds, respectively.

The effect on stage length and volume, as depicted in figures 3-44 and 3-45, is considerably less than what might be achieved through increased chamber pressure.

The effect of new structural materials on RDT&E, TFU and program costs, is illustrated in figures 3-46 through 3-49. In each of these figures, the cost variations for reductions in aluminum sheet/stringer structure weights have been shown together with a range of costs savings for both the honeycomb and the composite. Because these cost data include the new technology costs which are difficult to determine, a range of costs has been presented for the regions of weight reduction which are applicable to honeycomb and composite structures.

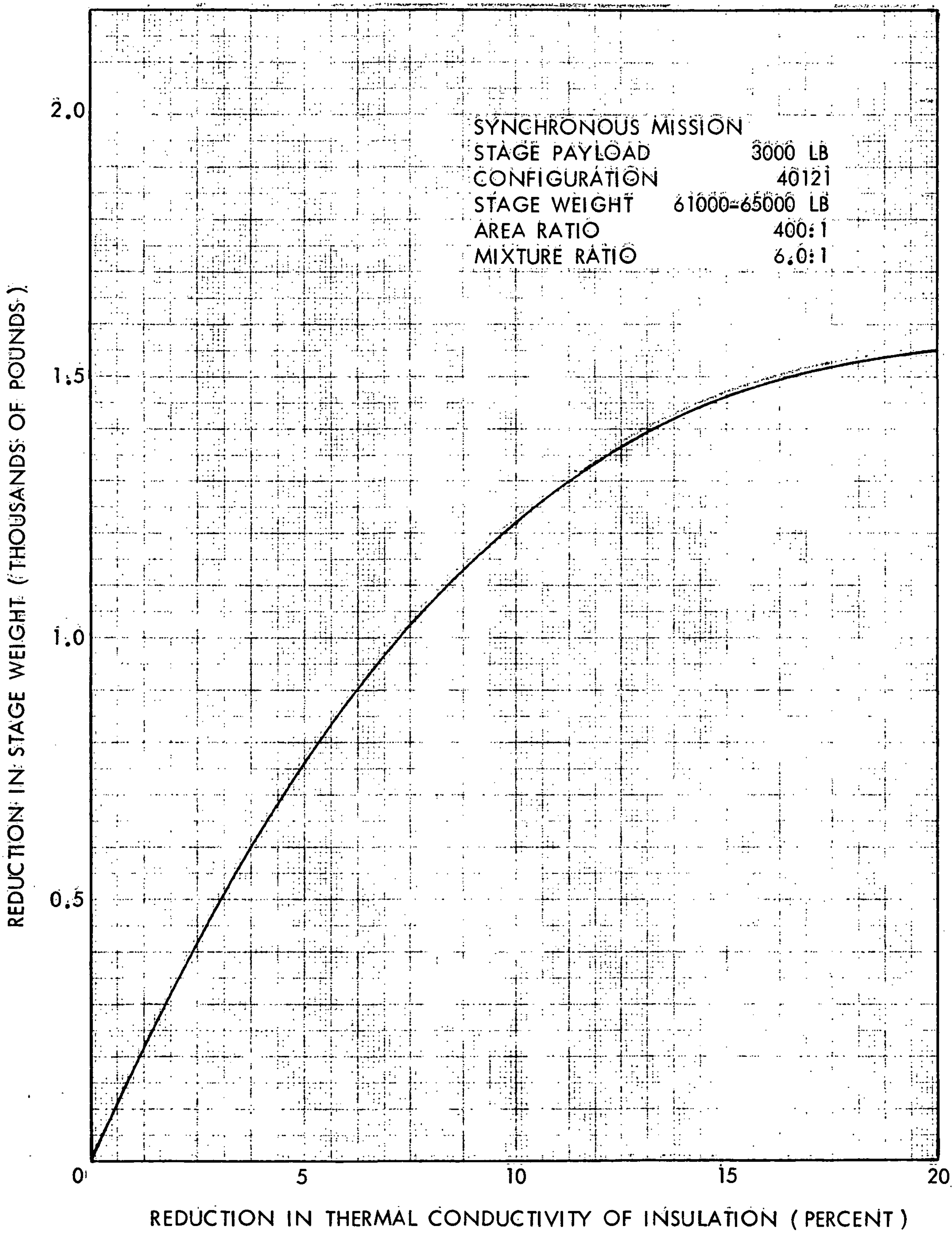

Figure 3-35. Variation of Stage Weight Due to Insulation Thermal Conductivity

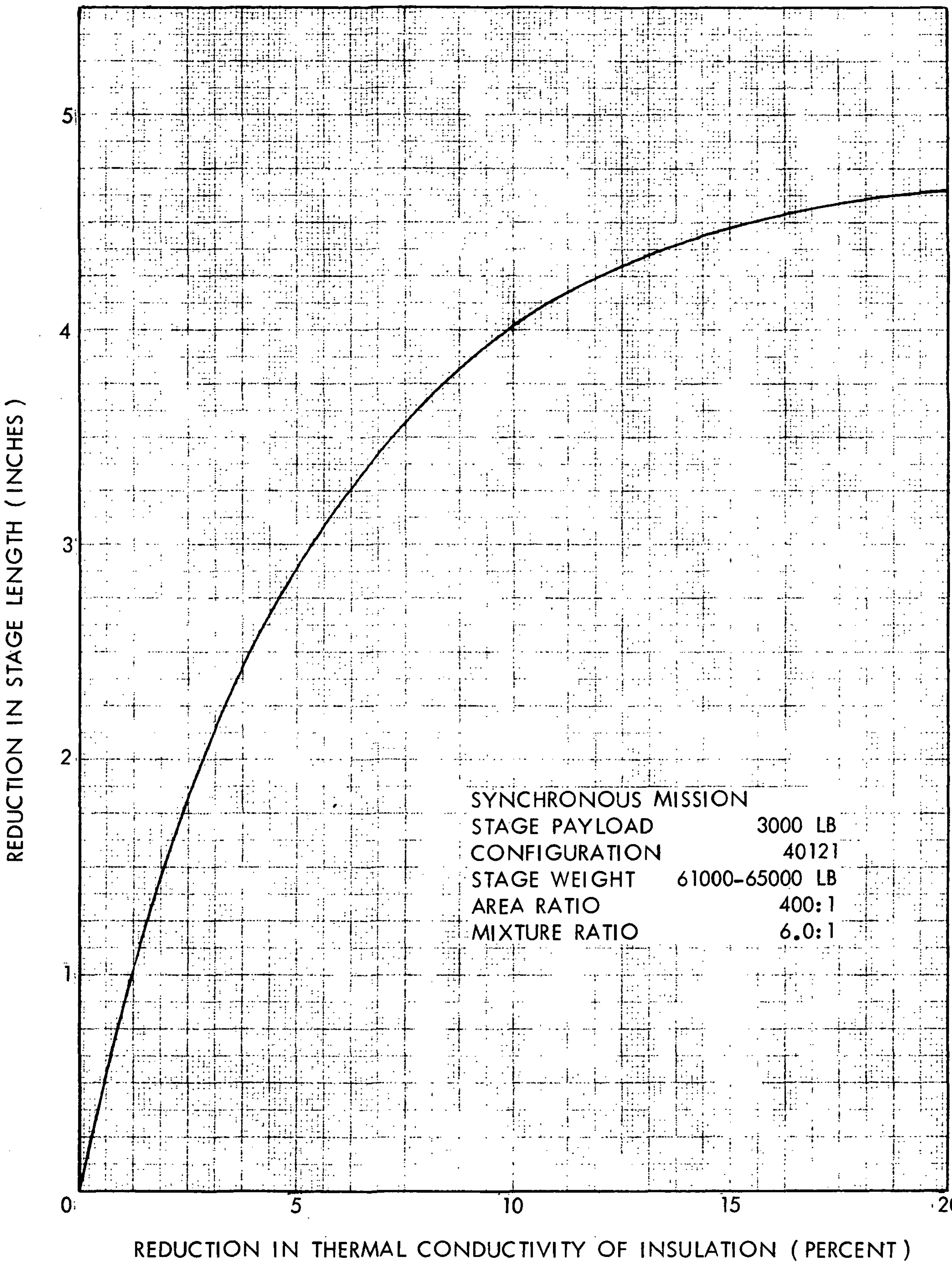

Figure 3-36. Variation of Stage Length Due to Insulation Thermal Conductivity

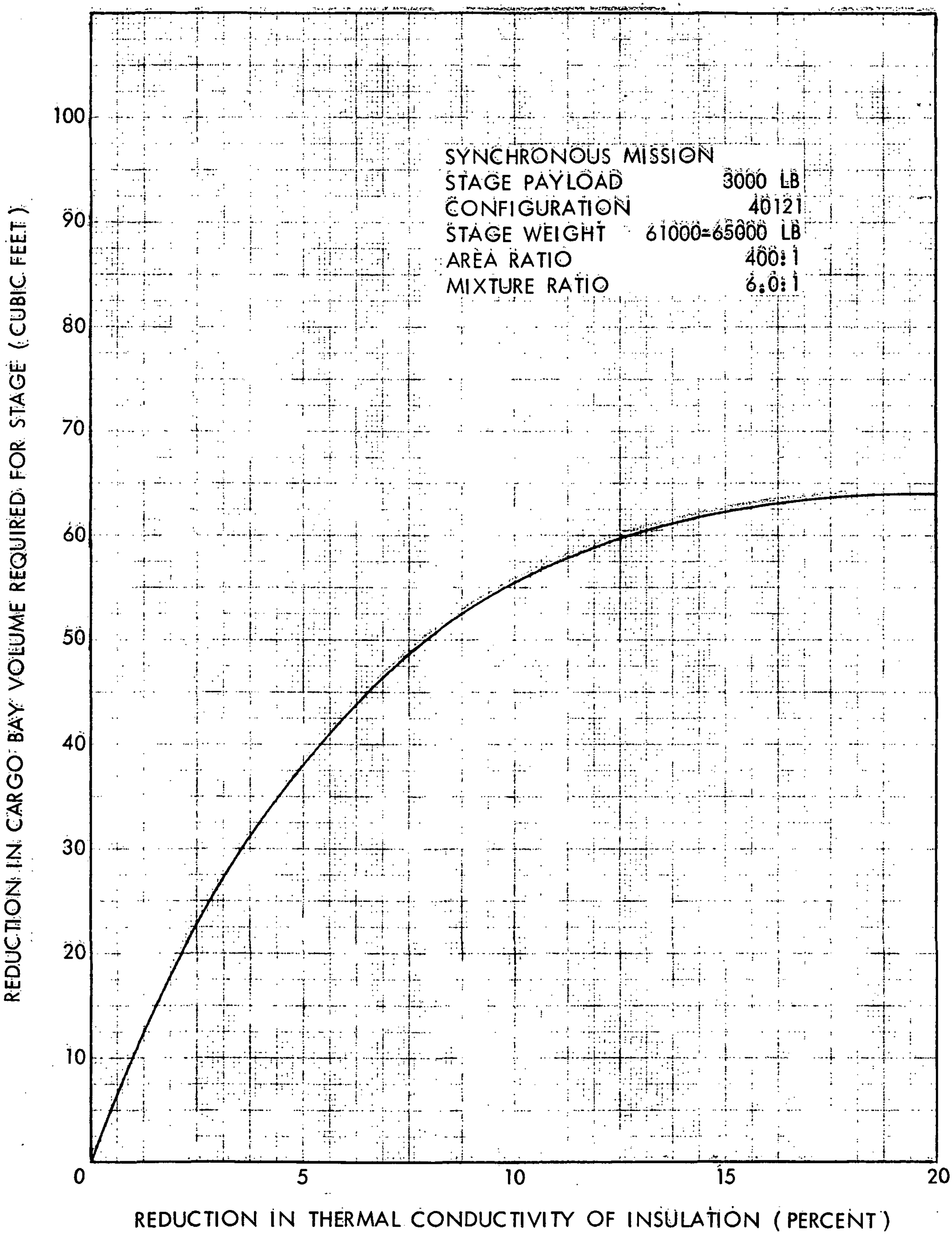

Figure 3-37. Variation of Stage Volume Due to Insulation Thermal Conductivity

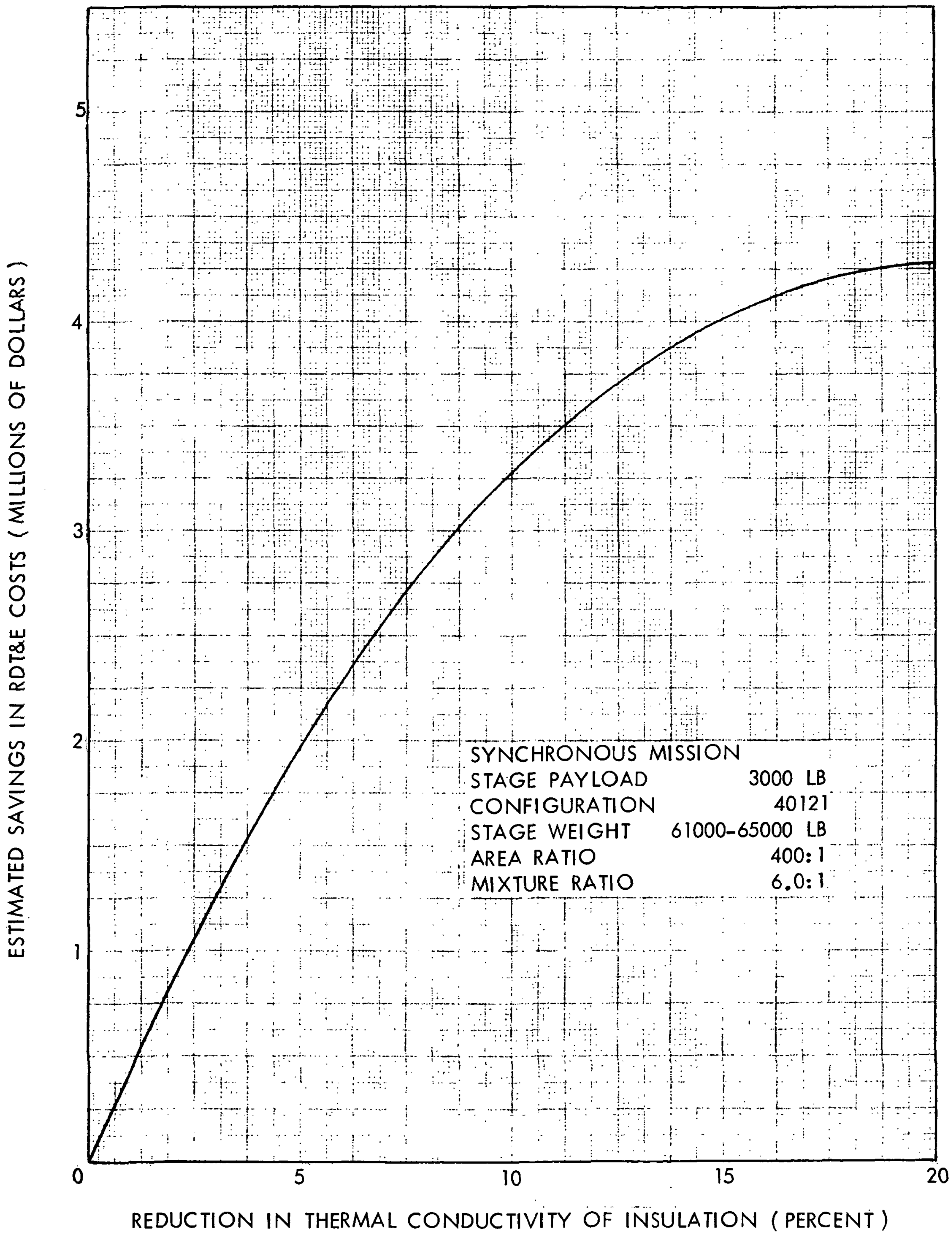

Figure 3-38. Variations of RDT&E Cost Due to Insulation Thermal Conductivity

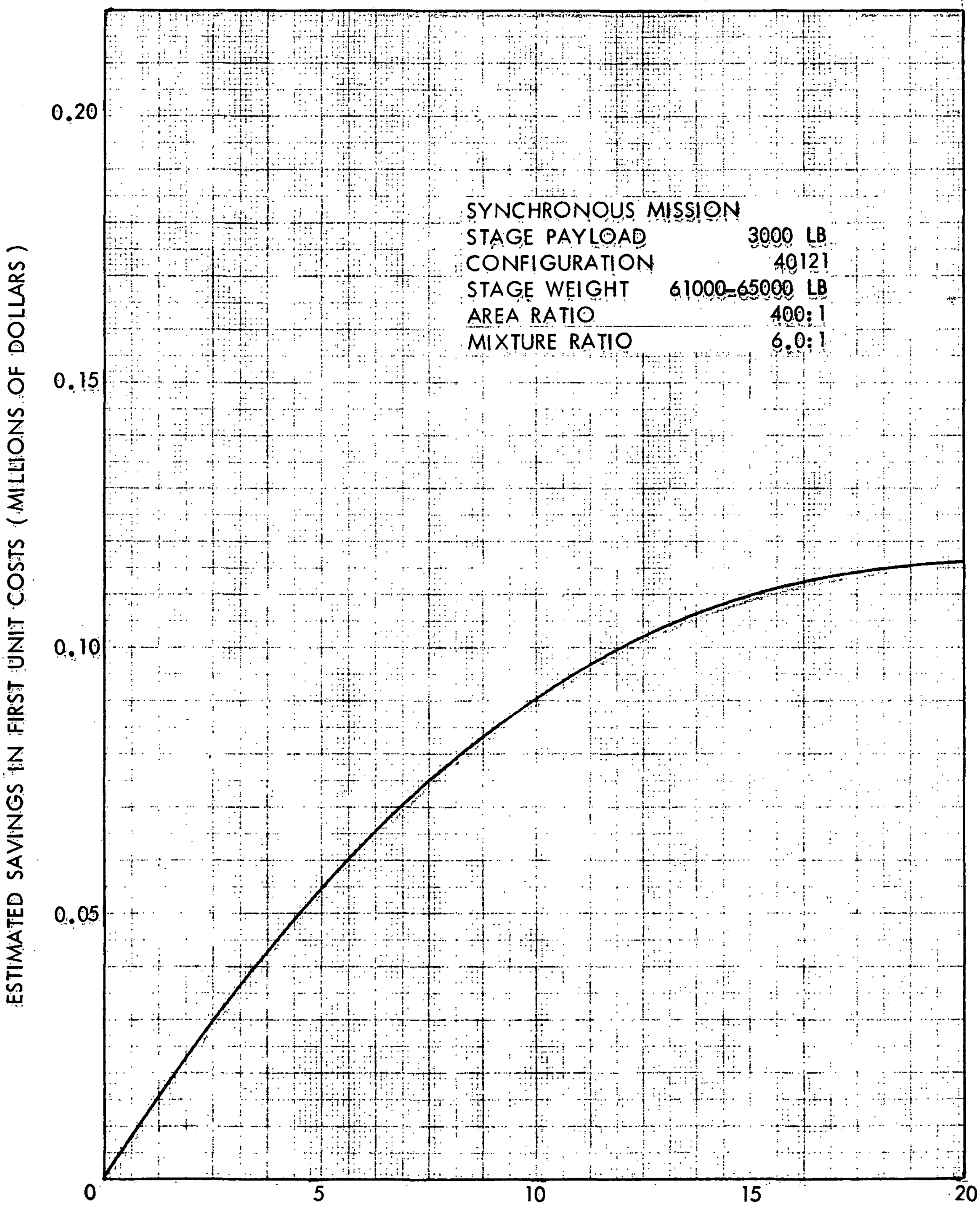

Figure 3-39. Variation of First-Unit Cost Due to Insulation Thermal Conductivity

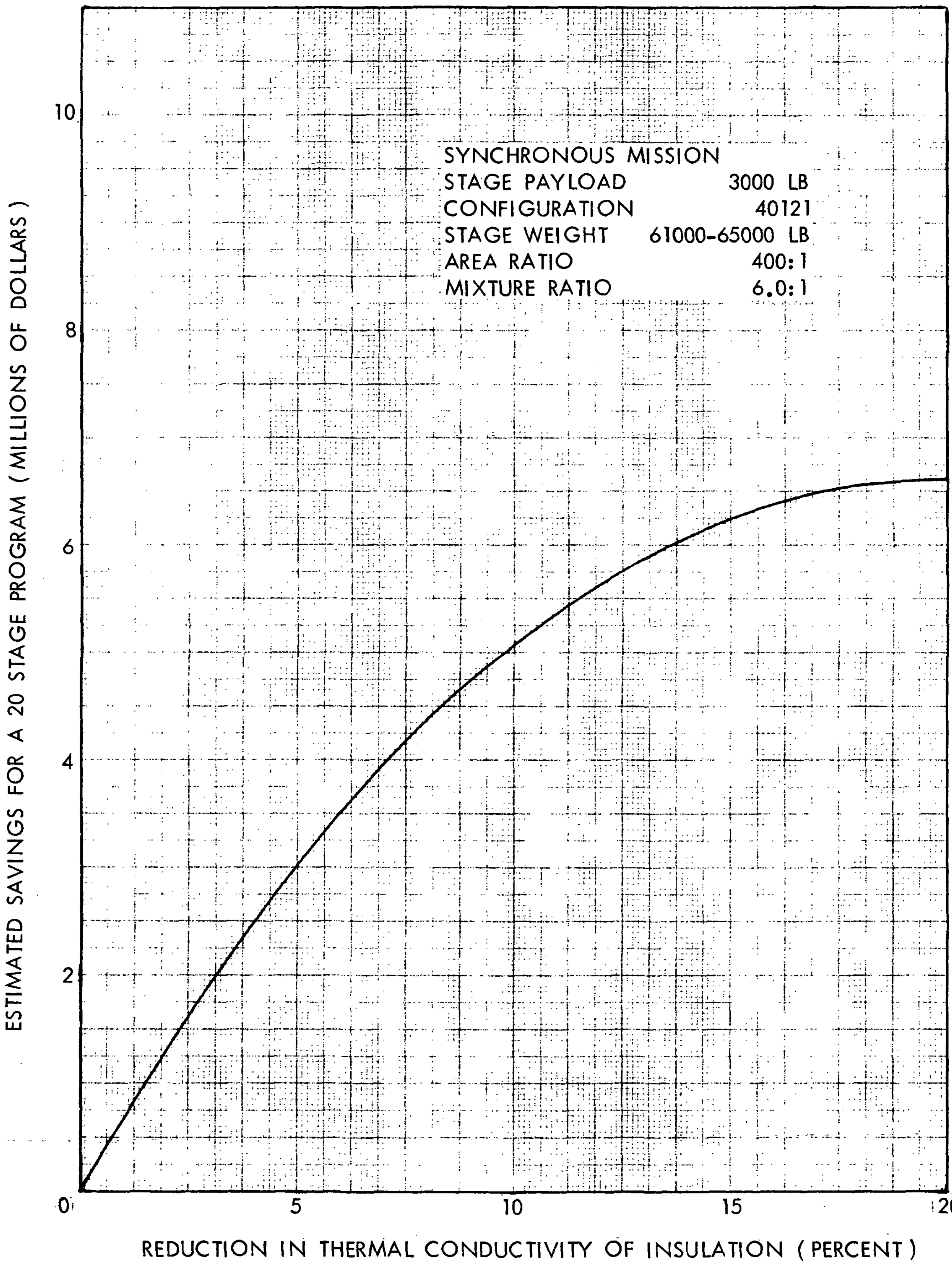

Figure 3-40. Variation of Program Cost Due to Insulation Thermal Conductivity

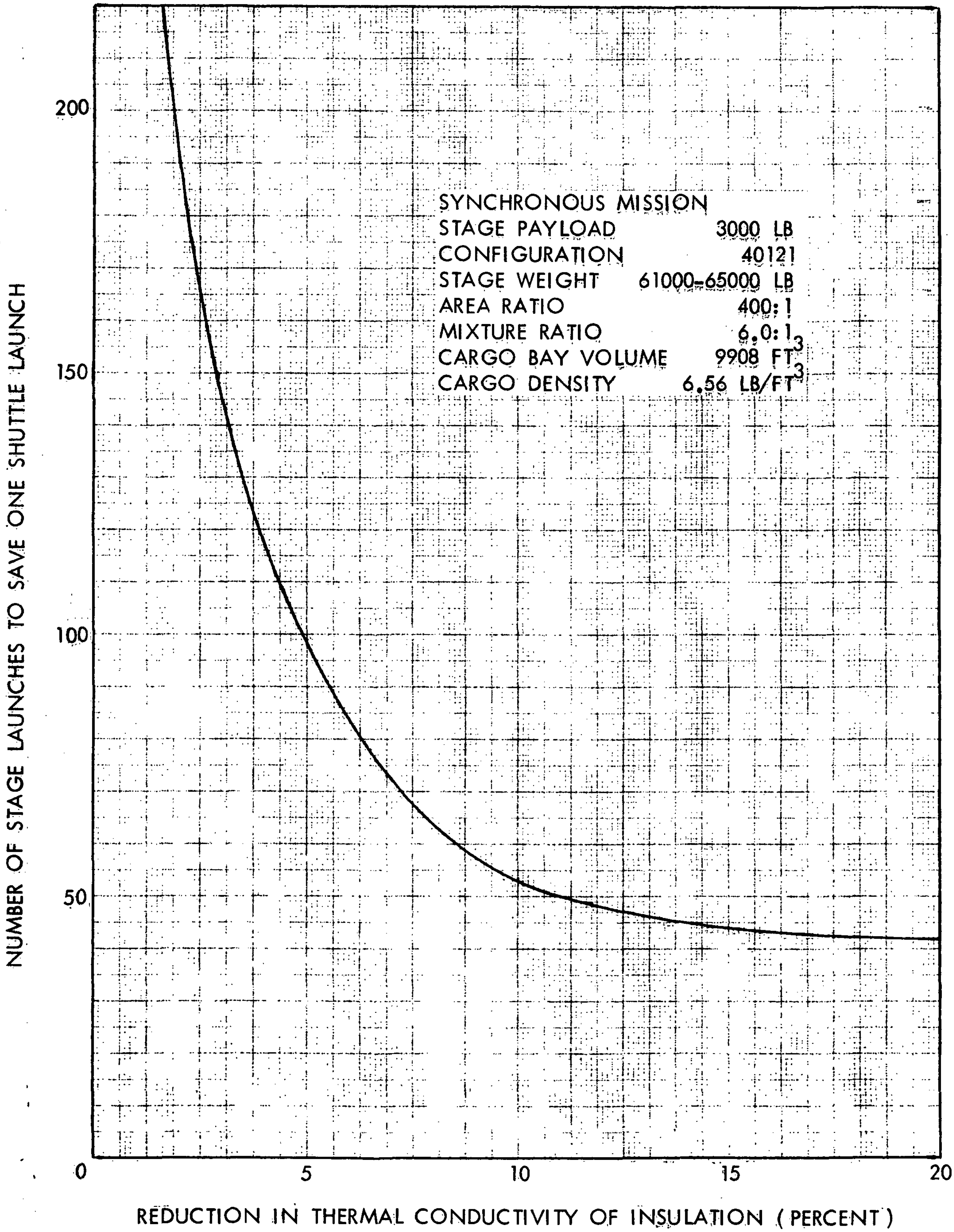

Figure 3-41. The Number of Stage Launches Required to Save One Shuttle Flight (Insulation Thermal Conductivity)

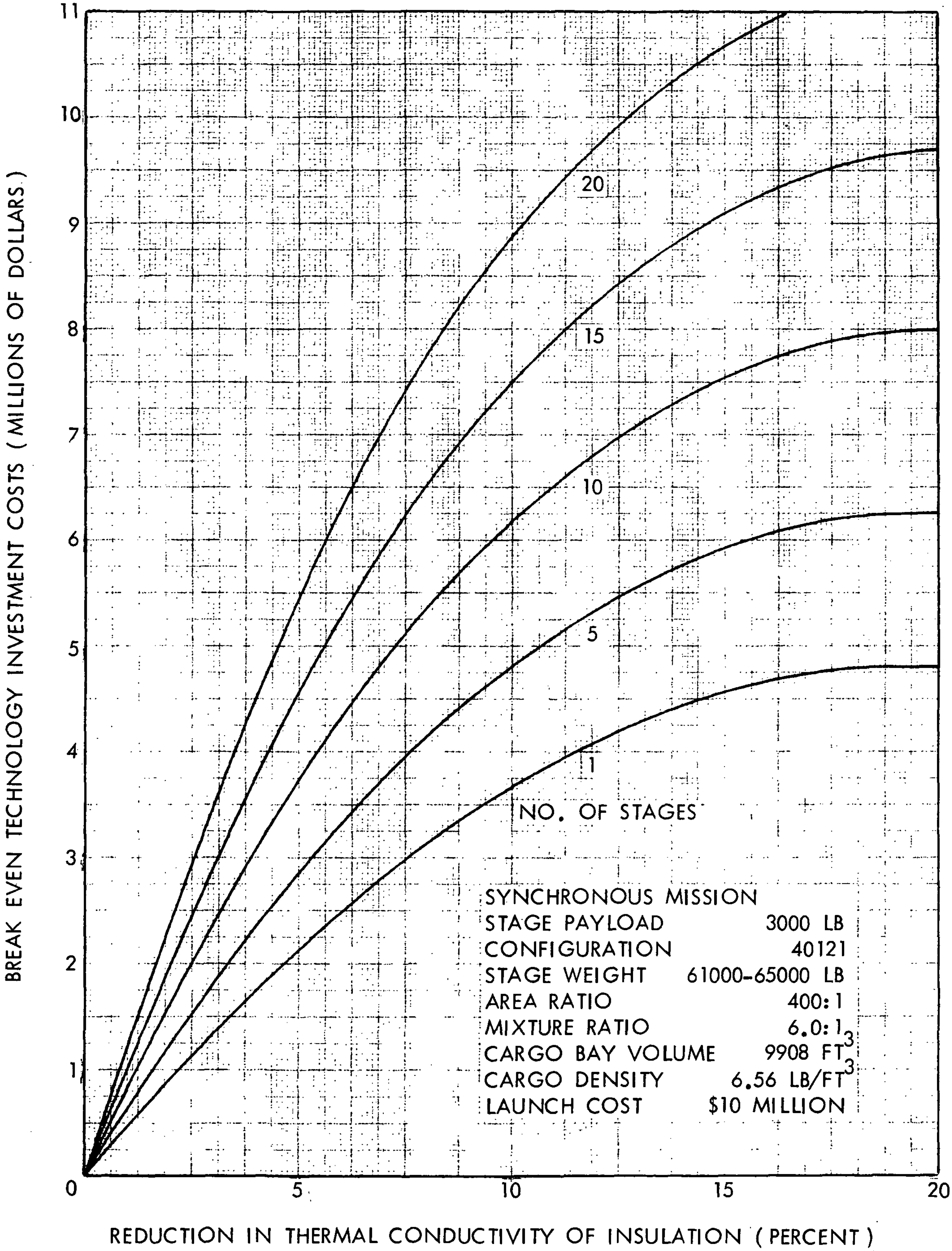

Figure 3-42. Break-Even Technology Costs for Insulation Thermal Conductivity

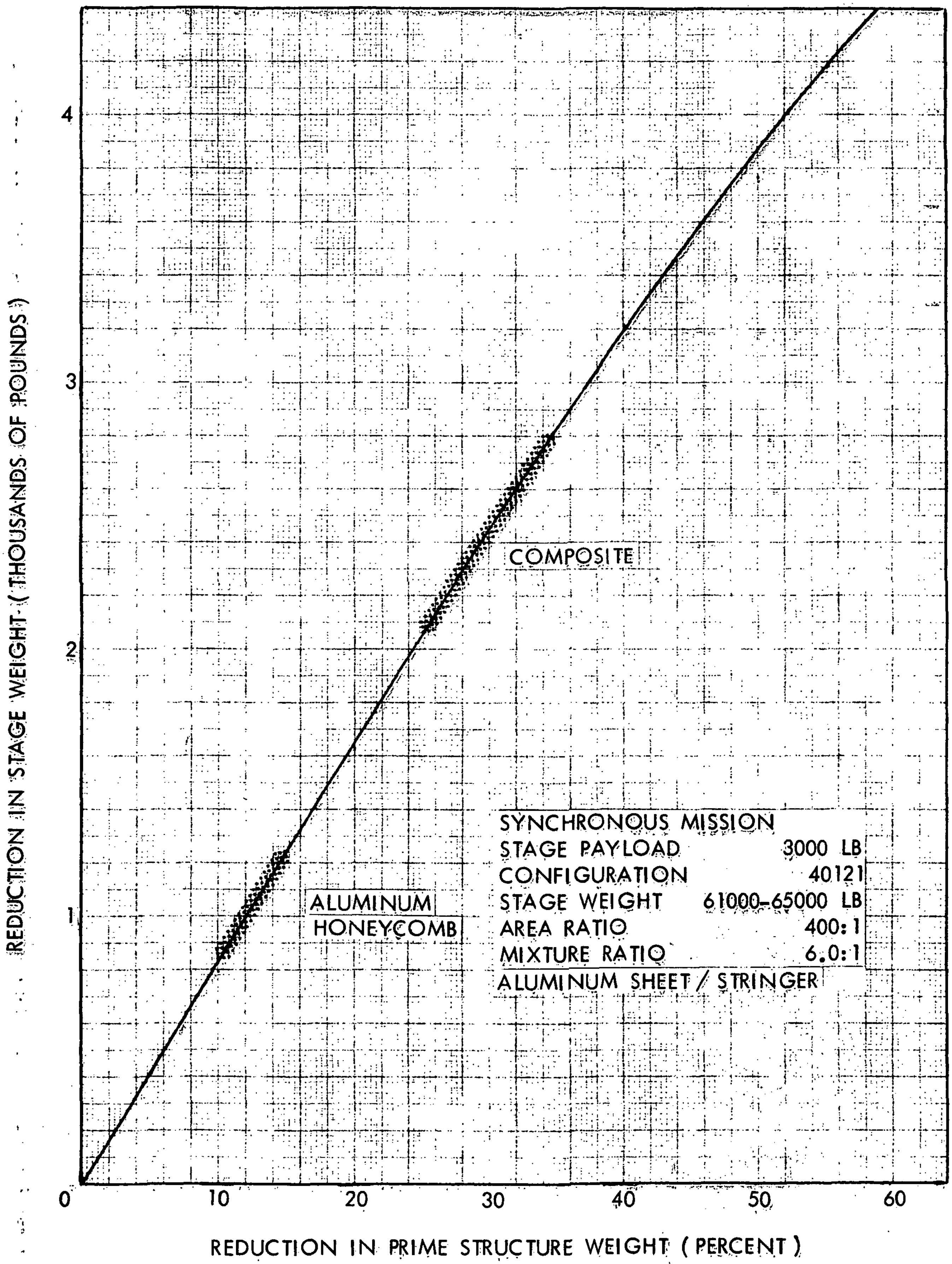

Figure 3-43. Variation of Stage Weight Due to Structural Weight

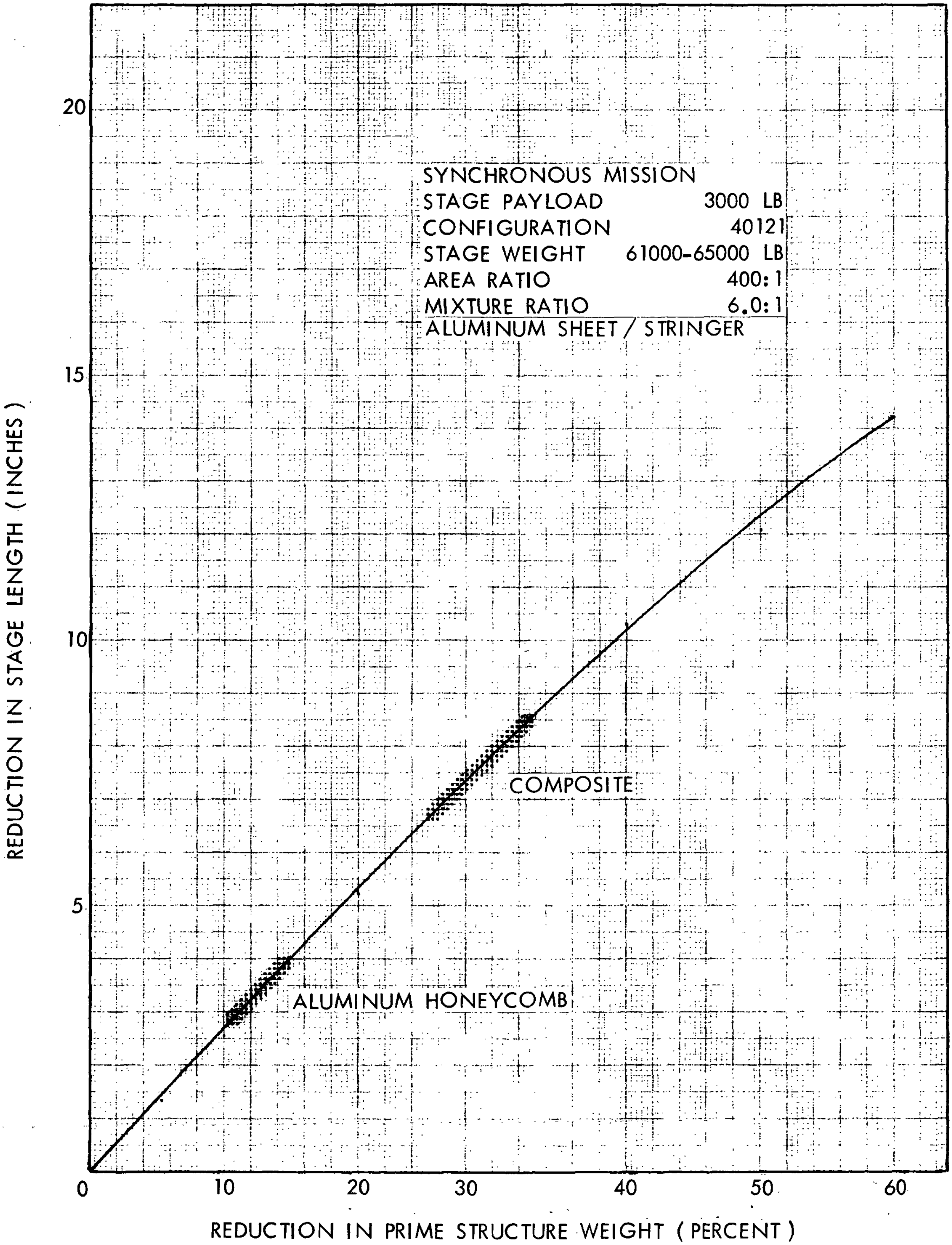

Figure 3-44. Variation on Stage Length Due to Structural Weight

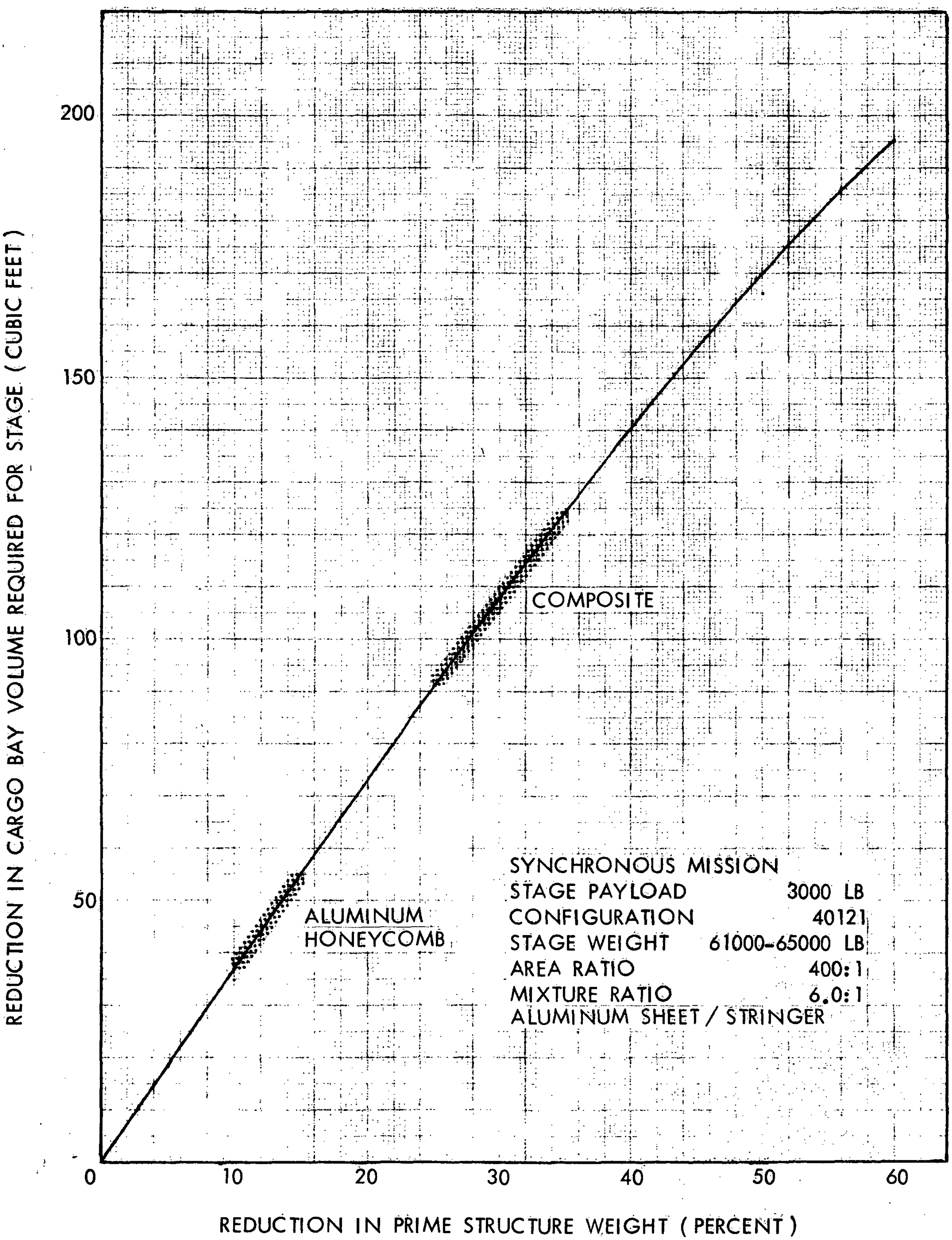

Figure 3-45. Variation of Stage Volume Due to Structural Weight

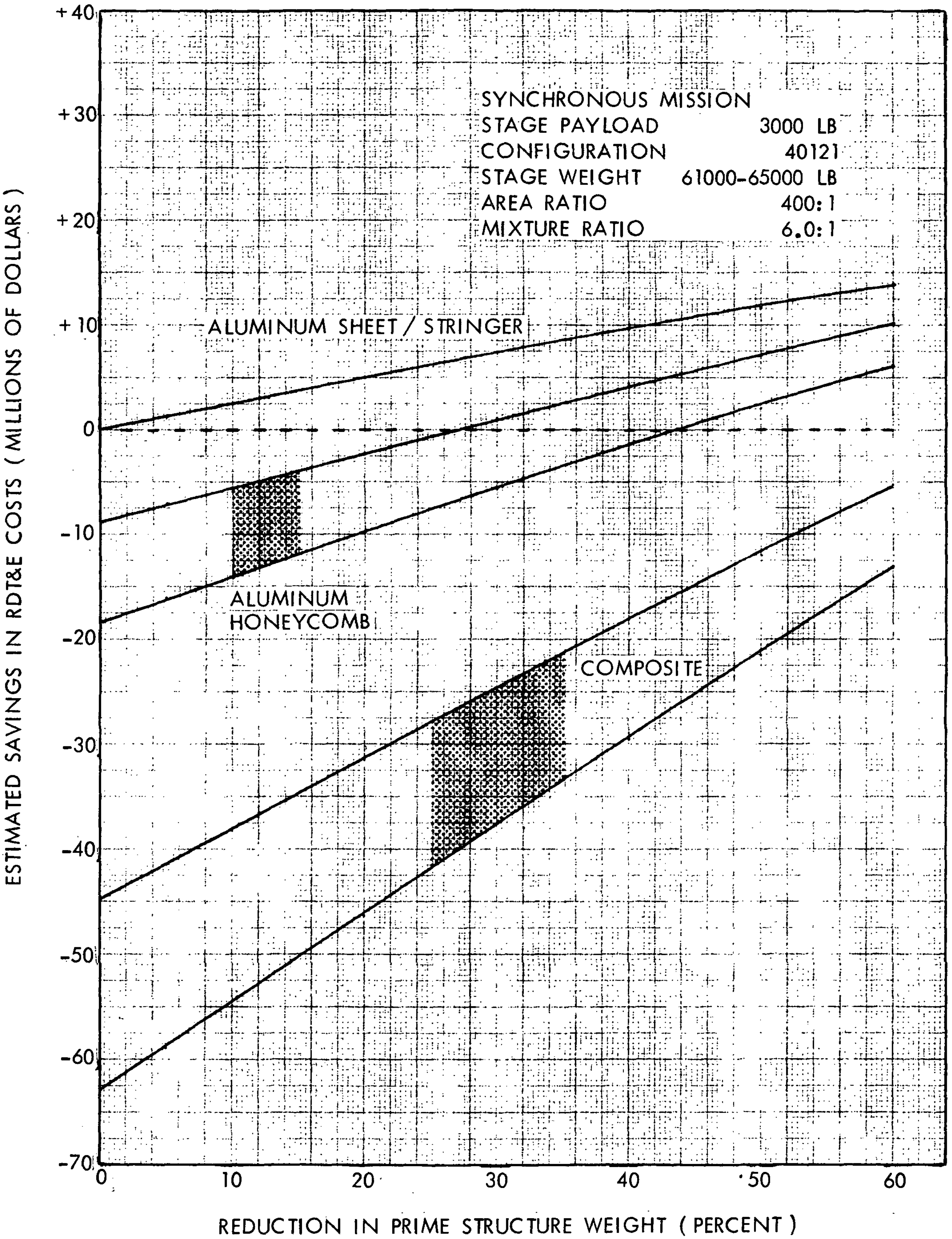

Figure 3-46. Variation in RDT&E Cost Due to Structural Weight

The RDT&E savings data, depicted in figure 3-46, show that a cost deficit would be incurred if either honeycomb or composite were used in place of the aluminum. The data also indicate that substantial weight reductions would be needed before a net savings could be incurred.

A similar trend is indicated for the investment costs, which are shown for the first unit in figure 3-47. In this instance, the cross-over points occur at an even larger percent of weight reduction than was found in the case of RDT&E costs.

Because of the large difference between the reductions in prime structure weight required to break even in RDT&E and investment (TFU) costs, the cross-over points for program cost savings will be dependent on the number of stages in the program. This is illustrated in figures 3-48 and 3-49, which show the savings estimated for a single-stage and a 20-stage program, respectively. For example, in order to break even with honeycomb, a 32 to 50 percent reduction in prime structure weight would be required in a single-stage program; while better than a 60 percent reduction would be required if a composite were used.

Although the weight savings obtained through the use of these new materials is too small to overcome their high costs, net savings might be obtained through the use of these new materials. For instance, it might be possible to use an existing engine if the stage's inert weight were reduced slightly. This would eliminate the high development costs associated with a brand new engine, and would probably more than make up the cost deficits associated with the advanced materials.

3.9 RELATIVE GAINS

A comparison of the five methods of improving a stage's mass fraction is depicted in table 3-12. This table shows the necessary improvements required in each discipline in order to recover the cost of developing the associated technology. The break-even technologies are presented for three different technology investment costs, and two different sizing programs.

For example, if $5 million were to be invested in technology development for a single-stage program, then in order to realize a net savings, either the chamber pressure would have to be increased to at least 1620 psi, or the engine weight would have to be reduced by 40 percent. Likewise, the break-even point for the engine nozzle expansion ratio would be 310:1. In the case of the propellant tank insulation thermal conductivity, the conductivity would have to be reduced an infinite percent before a $5-mission technology investment could be recovered. Thus, it would be impossible to obtain a net return on a $5-million investment to improve the insulation's thermal conductivity.

The technology break-even points presented in table 3-12 indicate that the two areas in which the maximum return can be obtained are: 1) increases in engine nozzle expansion ratio; and 2) reduction of miscellaneous subsystem weights. The areas, excluding new structural materials, which offer the least return are reductions in engine weight and insulation thermal conductivity.

3.10 CONCLUSIONS CONCERNING MASS FRACTION IMPROVEMENT

Table 3-13 summarizes the effects of the six methods of improving the stage's mass fraction.

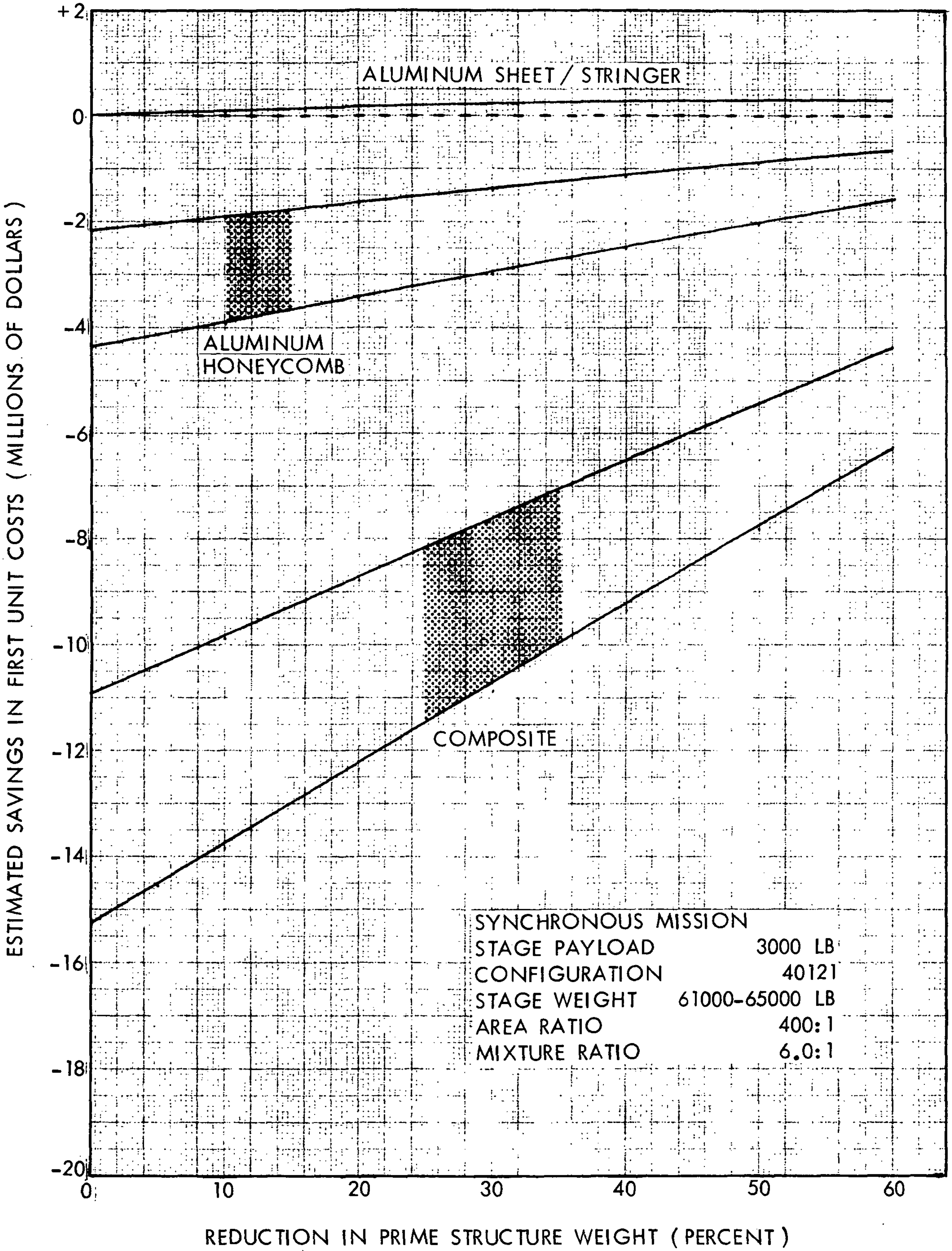

Figure 3-47. Variation of First-Unit Cost Due to Structural Weight

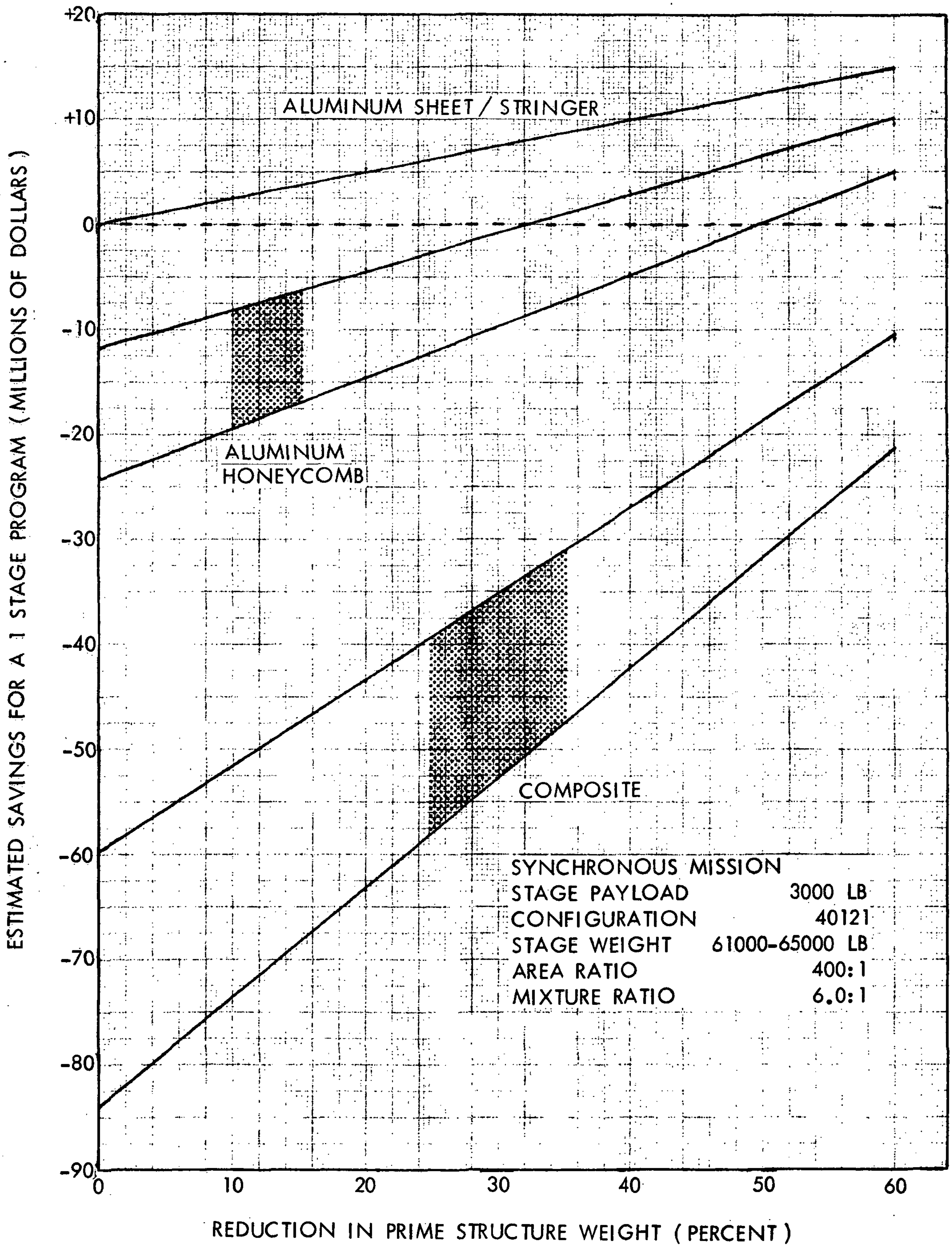

Figure 3-48. Variation in Single-Stage Program Cost Due to Structural Weight

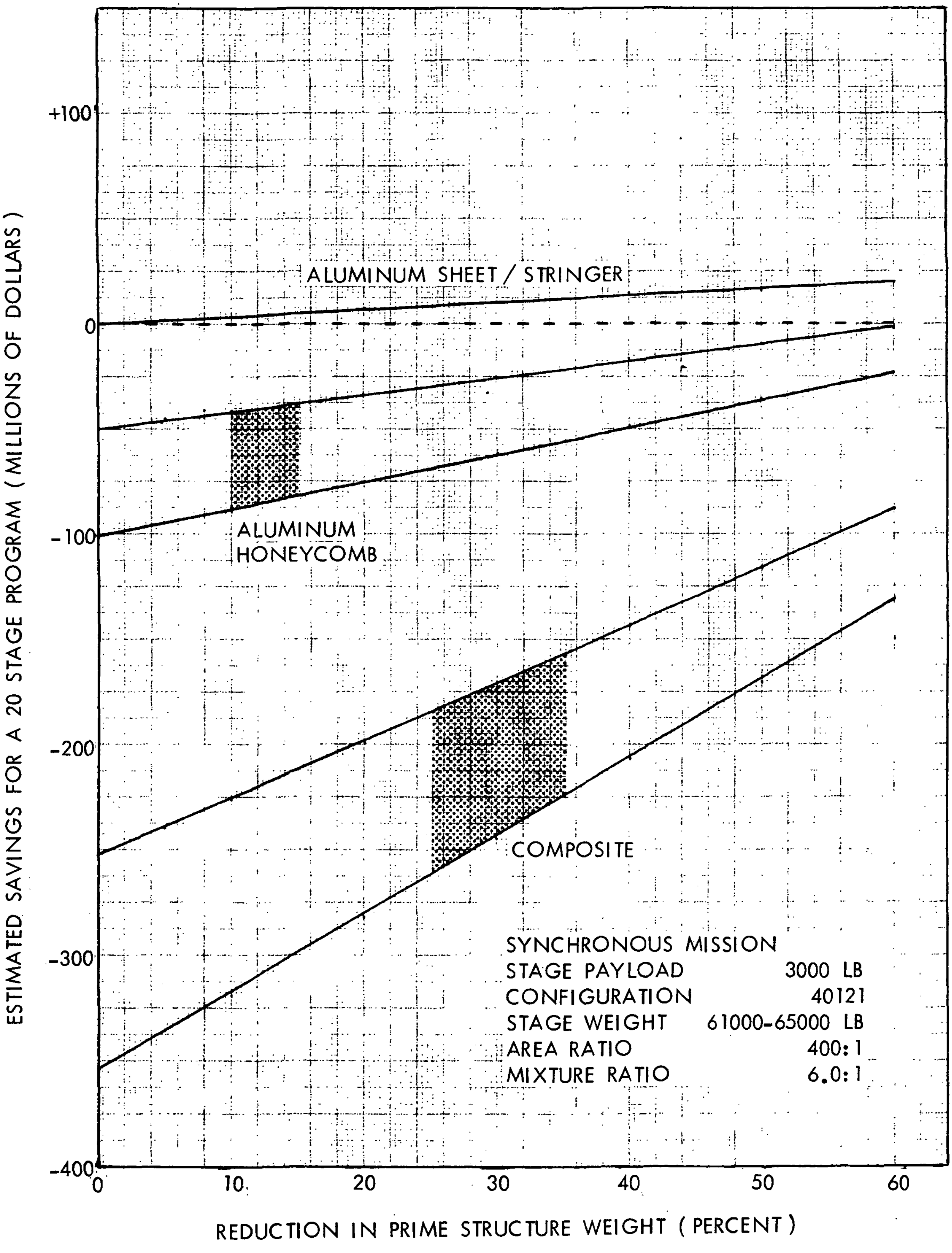

Figure 3-49. Variation in 20-Stage Program Cost due to Structural Weight

Table 3-12. Relative Break-Even Points

Technology Investment	No. of Flights	Chamber Pressure	Engine Weight	Area Ratio	Subsystem Weight	Thermal Conductivity
$1M	1	+900	-8%	+218:1	-1.0%	-2.25%
	20	+780	-3%	+206:1	-0.2%	-0.75%
$5M	1	+1620	-40%	+310:1	-4.0%	∞
	20	+950	-16%	+236:1	-2.0%	-4.50%
$10M	1	∞	∞	∞	-8.0%	∞
	20	+1280	-31%	+282:1	-4.0%	-12.75%

Table 3-13. Summary of Methods to Improve the Stages Mass Fraction

Area	Conclusion
Increase in Chamber Pressure	Total Net Savings are Marginal Resulting Decrease in Stage Length Important Only on Small Stages
Reduction in Engine Weight	Not Attractive from the Standpoint of Cost Savings
Increase in Engine Area Ratio	Yields Good Return on Investment Resulting Increase in Stage Length Important Only on Small Stages
Reduction in Miscellaneous Sybsystem Weight	Yields Good Return on Investment
Decrease in Tank Insulation Thermal Conductivity	Not Attractive from the Standpoint of Cost Savings
Reduction in Structure Weight	Probably Cost Deficit for New Materials Might be Attractive if New Materials Permit Use of Existing Engines

Section 4

TASK 2 - WORK ORDER 6., FUTURE PROPULSION CONCEPTS

4.1 GENERAL

At the present time, NASA has identified seven concepts which may be categorized as new horizons:

a. Atomic Hydrogen

b. Metallic Hydrogen

c. Metastable Activated Molecules

d. Compounds of Activated Helium

e. Activated Oxygen

f. Laser-Energized Fluids

g. Planet Atmosphere Reactions

The purpose of this task assignment was; 1) to estimate what propellant mass fraction might be associated with these concepts; and 2) to illustrate, quantitatively, what these concepts might mean in terms of reduced stage sizes and attainable velocities.

The following excerpts, taken from the "Progress Report on New Horizons for Propulsion", (9) briefly describe these new propulsion technologies:

a. <u>Atomic Hydrogen</u>

"The energy of recombination of atomic hydrogen to form molecular hydrogen is approximately 52 kilocalories per gram (kcal/gm). The energy could heat the resulting molecular hydrogen to about 4000 degrees Kelvin, producing a calculated specific impulse of about 1700 seconds on expansion. The method of production and stabilization of atomic hydrogen is to pass molecular hydrogen through a discharge tube causing the dissociation, and trapping the atomic hydrogen on a cold surface in the presence of a magnetic field of 70 to 100 kilogaus. The magnetic field theoretically should align the spin of the electrons in the atomic hydrogen in the same direction, thus preventing their recombination to form molecular hydrogen."

b. <u>Metallic Hydrogen</u>

"Hydrogen will convert to a close-packed, body-centered, cubic structure under extreme pressures, on the order of 1 megabar. In this condition, it is metallic and stores a large amount of energy; about 52 kcal/gm. Theoretically, it will be metastable at ordinary pressure; if so, the metallic hydrogen could be used in the form of a slurry in liquid hydrogen, similar to slush hydrogen now coming into practice. On proper "ignition" the metallic hydrogen would revert to normal hydrogen, releasing its energy and giving about 1700 seconds of specific impulse.

"Metallic hydrogen has never been isolated, although some workers claim that it has existed for a very short time in dynamic pressure apparatus, that is, in explosive-initiated experiments.

"It should be mentioned that metallic hydrogen is predicted to be a super-conductor, and there is a possibility that it will retain super-conductivity at a relatively high temperature. The high-pressure work to make it should have valuable outgrowth in solid-state physics and material properties."

c. <u>Metastable Activated Molecules</u>

"These are atoms or molecules which have been energized by electronic excitation, such as microwave discharge. In this condition, which is called the triplet state, they contain large amounts of energy which cannot be released by radiation, but must be released by recombination or reaction. The elements of interest are helium, which would contain 456.5 kilocalories per molecule (114 kcal/gm) in the excited state, neon and argon.

"The activated material might be used pure or in combination with a hydrogen or helium carrier. The specific impulse computed for pure He* is about 2800 seconds; for Ne* 1100 seconds; for Ar* 700 seconds. Excited neon is computed to have a half life of several years at 5 degrees K; active oxygen, discussed below, has a half life of several weeks at liquid air temperatures (minus 185 degrees C)."

d. <u>Compounds of Activated Helium</u>

"It may be possible to stabilize the activated species by the formation of compounds with other, normal species. Helium is usually considered to be a nonreactive element, but a number of workers have found it in compounds with elements such as mercury, iodine, sulfur, iron, platinum and paladium. There is also a basis for believing that activated helium may form weak combinations with normal helium.

"This molecule has been observed spectroscopically, and the data lead to a calculated specific impulse of 1920 seconds under conditions of expansion from 1,000 psi to 14.7 psi. Helides of lithium or hydrogen are also worthy of investigation."

e. Activated Oxygen

"Oxygen may be vibrationally excited to a high-energy state, and theoretically should be stable at liquid air temperature for a few weeks. While the energy availability does not approach that of helium, the possible use of activated oxygen to supplement existing hydrogen/oxygen propulsion systems is of considerable interest. The liquid hydrogen/liquid oxygen combination is estimated to have a specific impulse of 525 seconds on expansion from 1,000 psi to 1 atmosphere, which corresponds to 650 seconds under space conditions. This performance would be very beneficial for a single-stage orbit-to-orbit reusable tug where the velocity increment of approximately 28,000 feet per second means very high payoff from the increase in specific impulse.

"LeRC and JPL have only recently become aware of this concept of activated species, rather than atomic hydrogen as a potential propellant. Both LeRC and JPL are reviewing literature and making in-house surveys in order to define an appropriate course of action. A cursory search was conducted through recent aerospace abstracts and a good number of references to recent research in activated molecules were found, not surprising in view of the growth of laser science and technology in the last decade. For example, E. Muschlitz, University of Florida, published a paper in Science, February 1968, on metastable atoms and molecules, showing that helium has a number of metastable states with very high energy; i.e., 19 to 20 electron volts, equivalent to about 460 Kcal/atom. The lifetime of one of these species is stated to be very long. Neon and argon also have long-lived metastable states, with energy about 370 Kcal/atom (Ne) and 265 Kcal/atom (Ar).

"In sum, the use of activated metastable species for propulsion appears to be a concept with potential for large performance gains over the best normal propellants; viz., more than an order of magnitude in energy content, and four-fold increase in specific impulse. The research to establish such new capability will benefit greatly from laser research, because this also involves understanding of activated states."

f. Laser-Energized Fluids

"The essence of this technique is the use of the energy in laser beams to heat a working fluid to yield thrust. The basic virtue of this energy source is that it is not limited by the energy available in the chemical bond, and it need not be located on the vehicle. The working fluid can be brought to a condition whereby it can produce extremely high thrust per unit mass. Further, lasers can react with solids to produce plasmas with very high expansion velocities, i.e., 10^7 centimeters per second. These offer the possibility for momentum exchange to produce thrust by a mode different from the expansion of a heated working gas. In this mode, the selection of the working substance would not be circumscribed by the need for isentropic expansion of gases; thus solids, such as lithium hydride, as well as liquids or gases could be considered for propulsive elements.

"One concept of laser-powered propulsion is beaming the laser from a fixed station, which could be on earth or on a satellite, to a vehicle. The energy is absorbed in a working fluid, typically hydrogen, and converted into internal energy. The specific impulse computed for this combination is limited by the heat transfer to the nozzle and combustion chamber walls, and, of course, by the power for the laser source. Some engineering studies have indicated that material temperatures can be maintained on the order of 5000 degrees R, with the corresponding maximum specific impulse on the order of 2500 seconds. If techniques for producing materials with high reflectivity can be perfected, wall temperatures as low as 2000 degrees R may be maintained, with specific impulses of approximately 5000 seconds.

"Water and methane have been considered for propellants. These will be dissociated and will produce specific impulses of 1250 and 1600 seconds, respectively, when the maximum temperature gives a 2500-second hydrogen specific impulse. The advantage from use of other propellants would be reduced volume due to their greater density.

"A propulsion system based on the interaction of laser readiation with a surface was analyzed in detail (AVCO). The propulsive mechanism involved the production of high temperature plasma with strong laser absorption. The specific impulse exceeded 1000 seconds.

"It seems feasible to make very low thrust laser-powered systems, to meet mission requirements such as station keeping or fine-pointing attitude control rockets. The source could be a sun-pumped laser on the satellites, which would produce plasma with a specific impulse at 1000 seconds or more. The propellant could be a solid such as carbon, aluminum, teflon, lithium, tungsten, etc."

g. Planet Atmosphere Reactions

"The atmosphere of Mars, Venus, and Jupiter will produce high-energy reactions with selected propellants. Thus CO_2, the main constituent of Mars' atmosphere, will react with beryllium to produce 11 kcal/gm; with boron, 7.5 kcal/gm. The nitrogen in Venus' atmosphere reacting with boron gives 5.8 kcal/gm; with beryllium 5 kcal/gm. These can be compared to one of the most energetic bipropellant reactions now available for propulsion: hydrogen fluoride, approximately 3 kcal/gm of combined propellant. The atmosphere of Jupiter is mostly hydrogen and has the potential for reaction with oxidizers such as oxygen, fluorine or ammonium perchlorate.

"These potentialities were studied at JPL beginning in 1965, and they are continuing to receive attention. It appears that air-breathing power systems operating in the Martian or Venusian atmospheres may result in reductions in the mass launch from earth, in comparison with other chemical power systems. They also may be competitive with nuclear systems for operating lifetimes of the order of weeks or less.

"JPL is continuing to analyze the potential applications of air-breathing systems and to rate them against competing rocket, nuclear, or chemical power systems."

The results of the analyses performed under this work order are presented in the remainder of this subsection.

4.2 PERFORMANCE AND SIZING CARPET PLOTS

In order to show the maximum attainable ideal velocities and various stage sizing parameters (e.g., PL/W_S, W_S) as functions of specific impulse and propellant mass fraction; the following carpet plots were prepared:

a. Maximum ideal velocities attainable with a single stage (figure 4-1).

b. Maximum ideal velocities attainable with two equal-size stages (figure 4-2).

c. Maximum attainable ideal velocities for a 50,000-pound stage with a 1500-pound payload (figure 4-3).

d. Stage sizing for a round-trip synchronous-orbit mission with the ratio of payload-to-stage weight as the dependent parameter (figure 4-4).

e. Stage sizing for a 3000-pound payload round-trip synchronous-orbit mission, with stage weight as the dependent parameter (figure 4-5).

These charts may be used to show how new propulsion concepts extend the range which is accessible by means of propulsion (i.e., the velocity plots) and/or the reduction in stage sizes required to accomplish a given mission--provided the user is able to estimate the propellant mass fraction associated with the propulsion concepts. A part of this task was to prepare conceptual designs to be used in estimating the packaging efficiency for various concepts. Because of the limited amount of time remaining in the study, only two concepts. were investigated--metallic hydrogen and activated helium. The activated helium was selected instead of activated neon or argon because it results in a stage with a lower mass fraction--i.e., it will give a lower bound on the mass fraction of stages using activated/metastable species. The reason helium packages with the least efficiency is because it has the lowest density, the lowest critical temperature and the lowest heat of vaporization. That is, it requires relatively large tanks and has a low capacity for absorbing heat; thus requiring more thermal protection.

The activated oxygen/hydrogen system was not examined, although it does appear to be very attractive. This is because it would be necessary, at this time, to assume that the activative.oxygen would have the same physical properties as normal oxygen. Thus, it would have a predicted mass fraction the same as a normal LOX/Hydrogen stage.

4.3 SIZING DATA AND ASSUMPTIONS

Both the metallic hydrogen and the activated helium behave as monopropellants; therefore, the monopropellant option of Chrysler Upper Stage Sizing Evaluation Routine (CUSSER) was used to estimate the propellant mass fractions

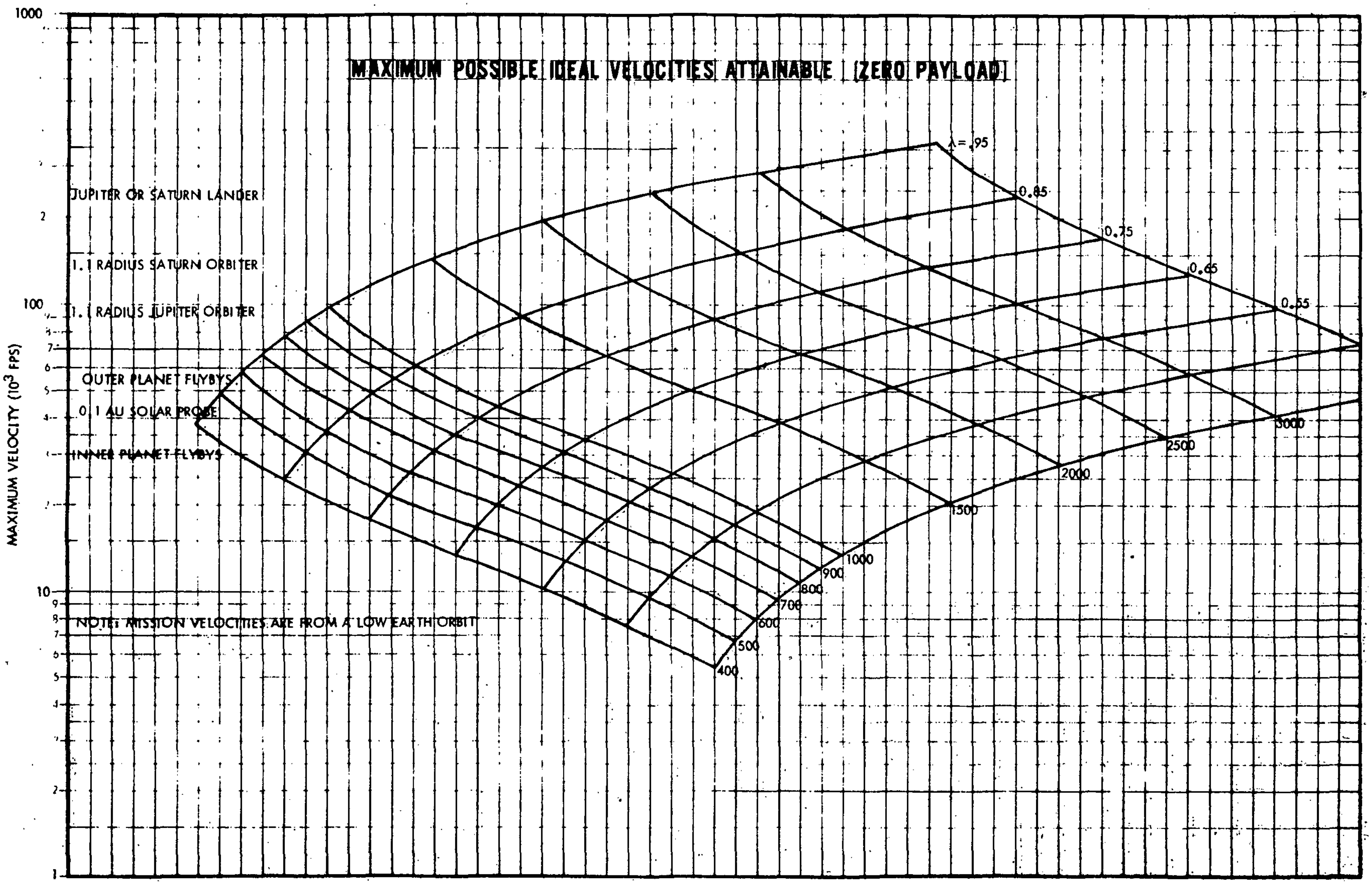

Figure 4-1. Maximum Ideal Velocities Attainable (Single Stage)

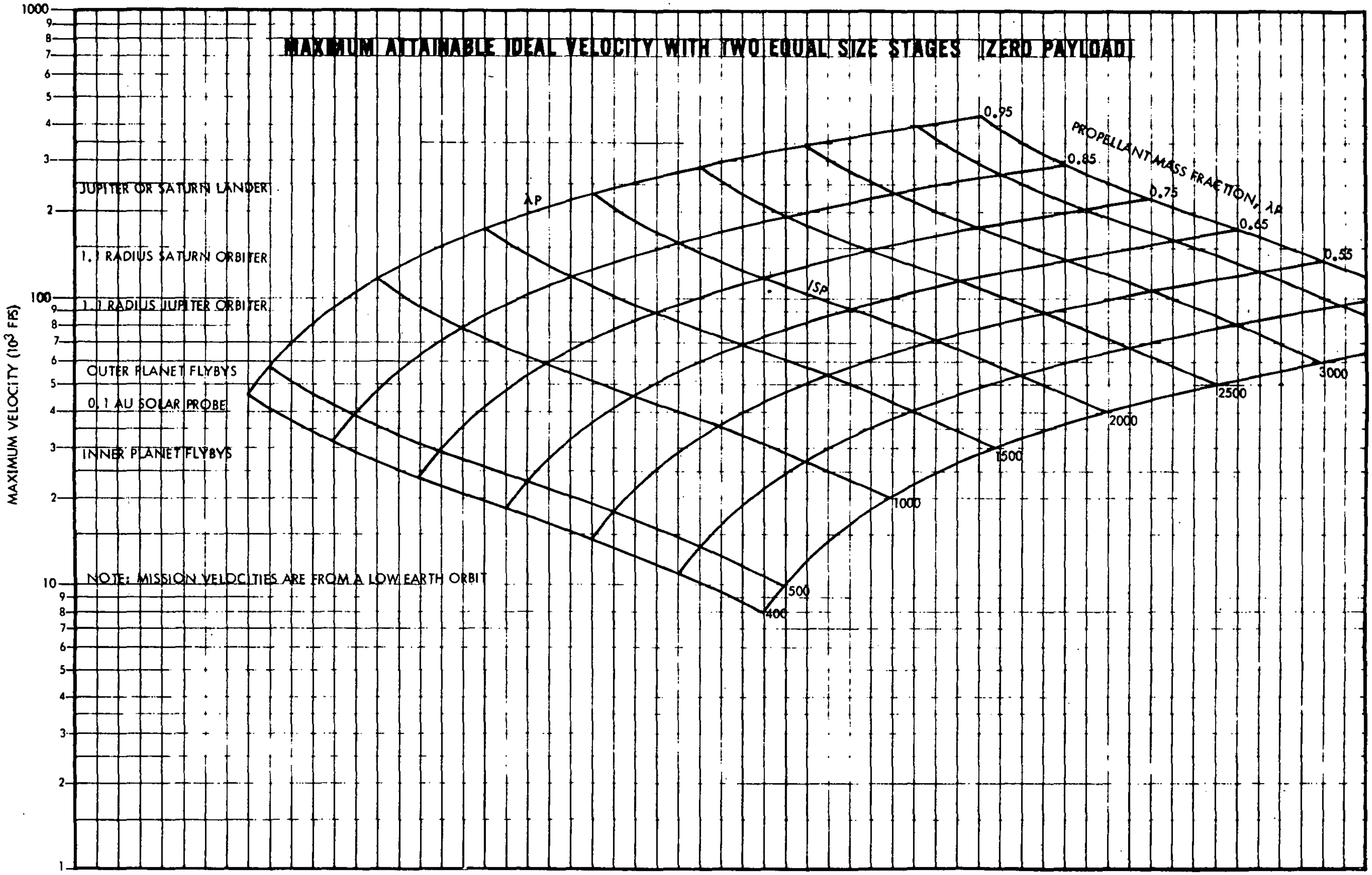

Figure 4-2. Maximum Ideal Velocities Attainable (Two Stages)

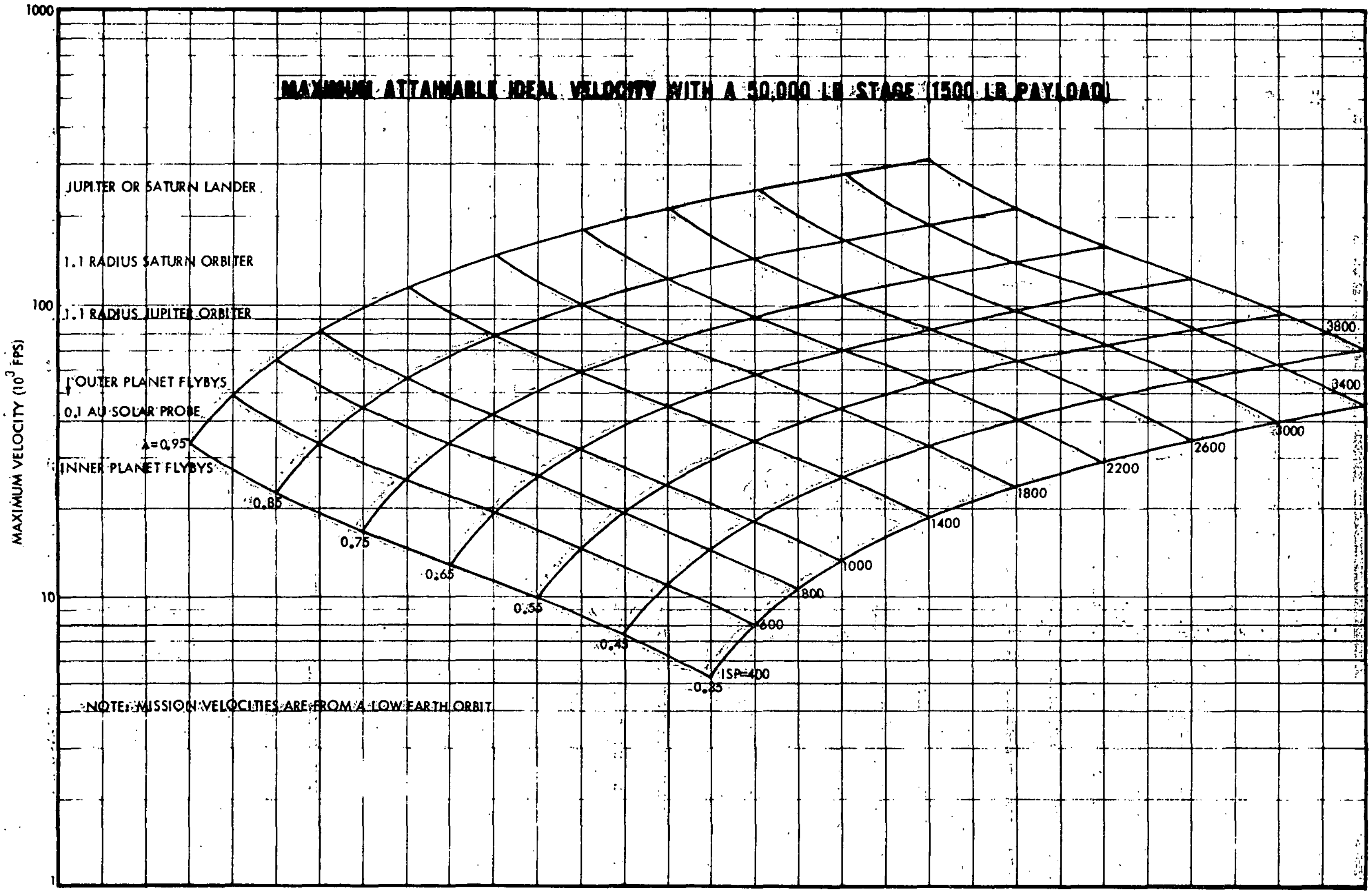

Figure 4-3. Maximum Ideal Velocities Attainable (50,000-lb Stage, 1500-lb Payload)

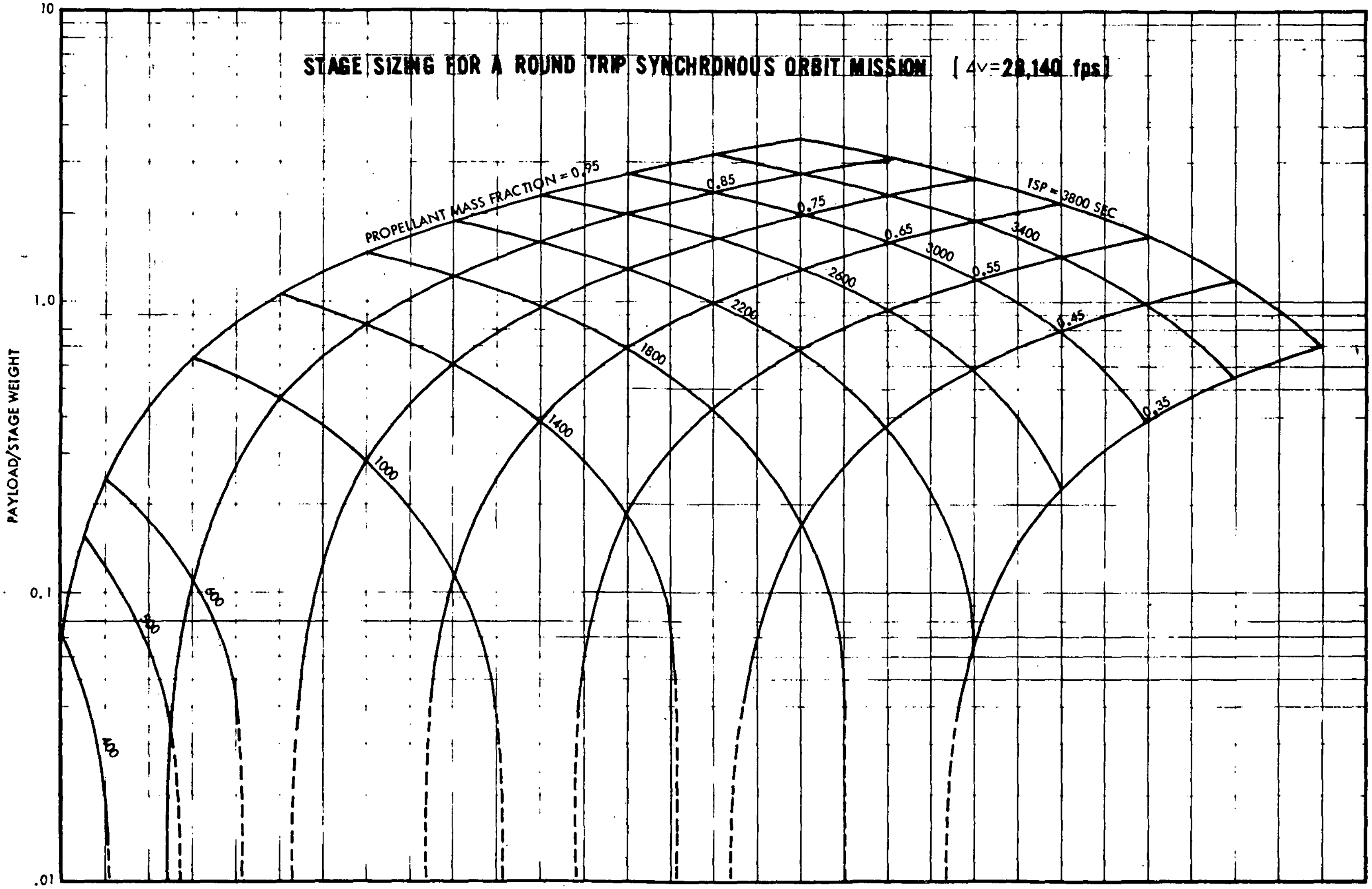

Figure 4-4. Round-Trip Synchronous-Orbit Mission Stage Sizing
(Payload-to-Stage Weight Ratios)

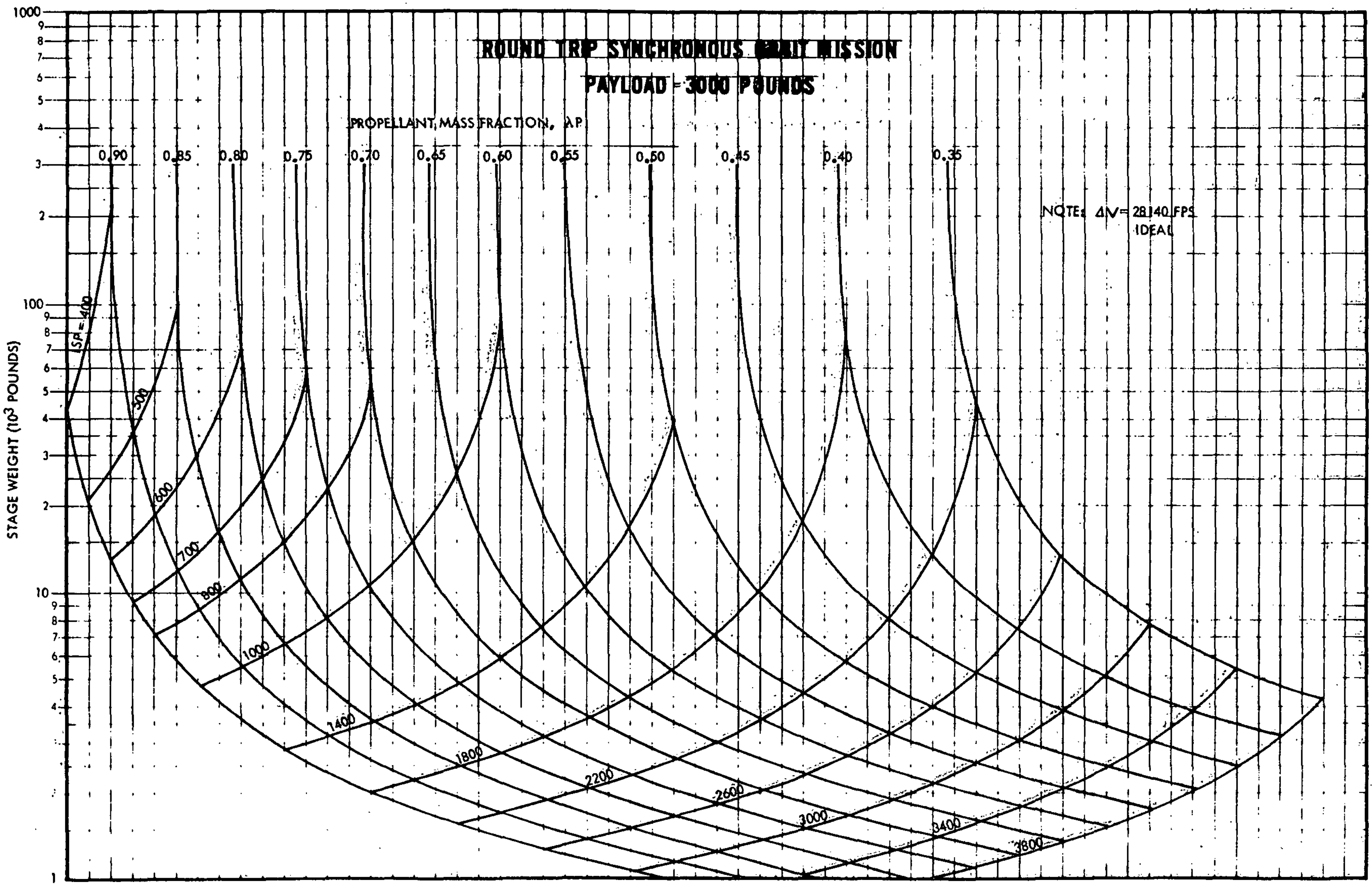

Figure 4-5. Round-Trip Synchronous - Orbit Mission Stage Sizing for a 3000-lb Payload (Stage Weights)

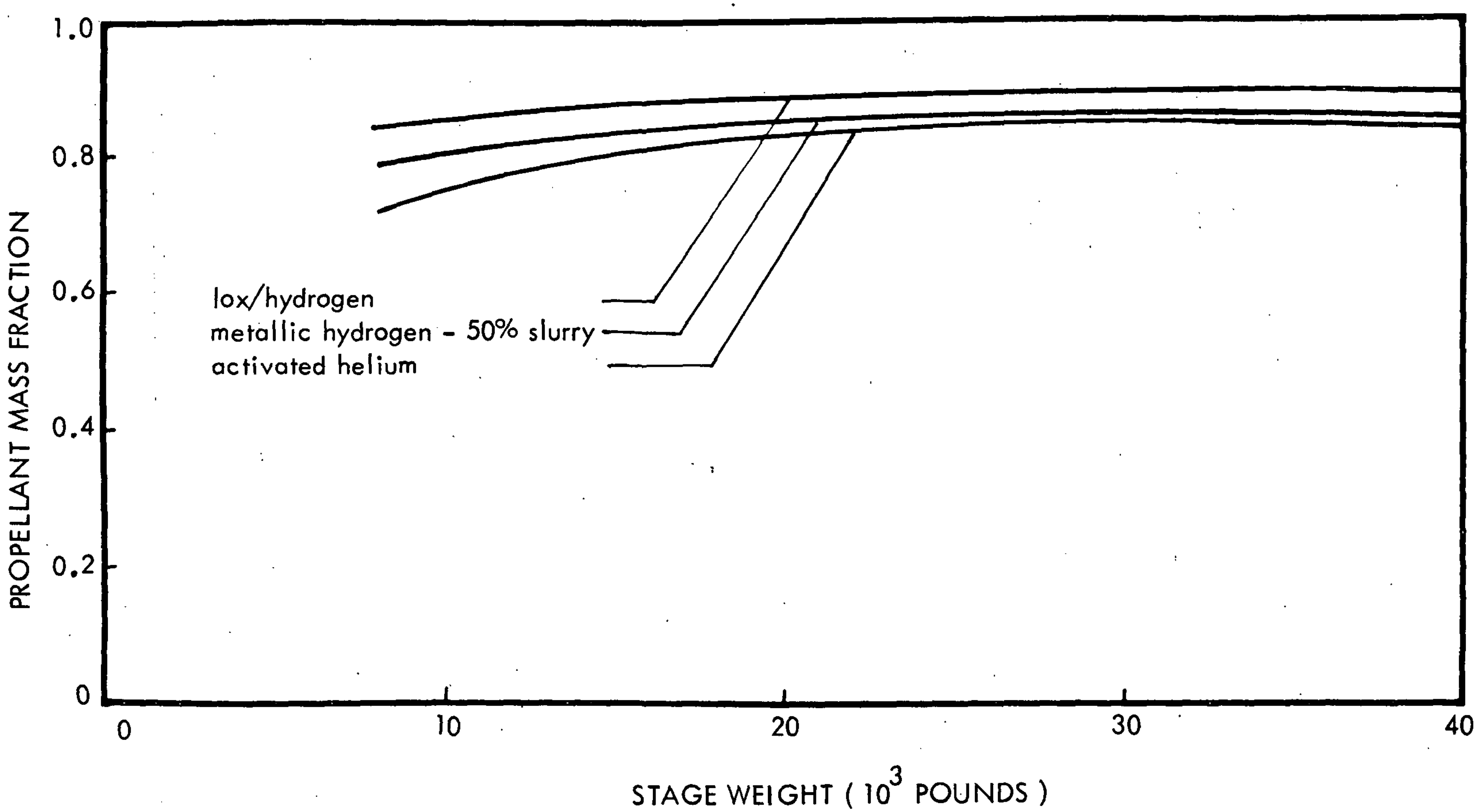

Figure 4-6. Effective Propellant Mass Fractions For Short Duration Missions

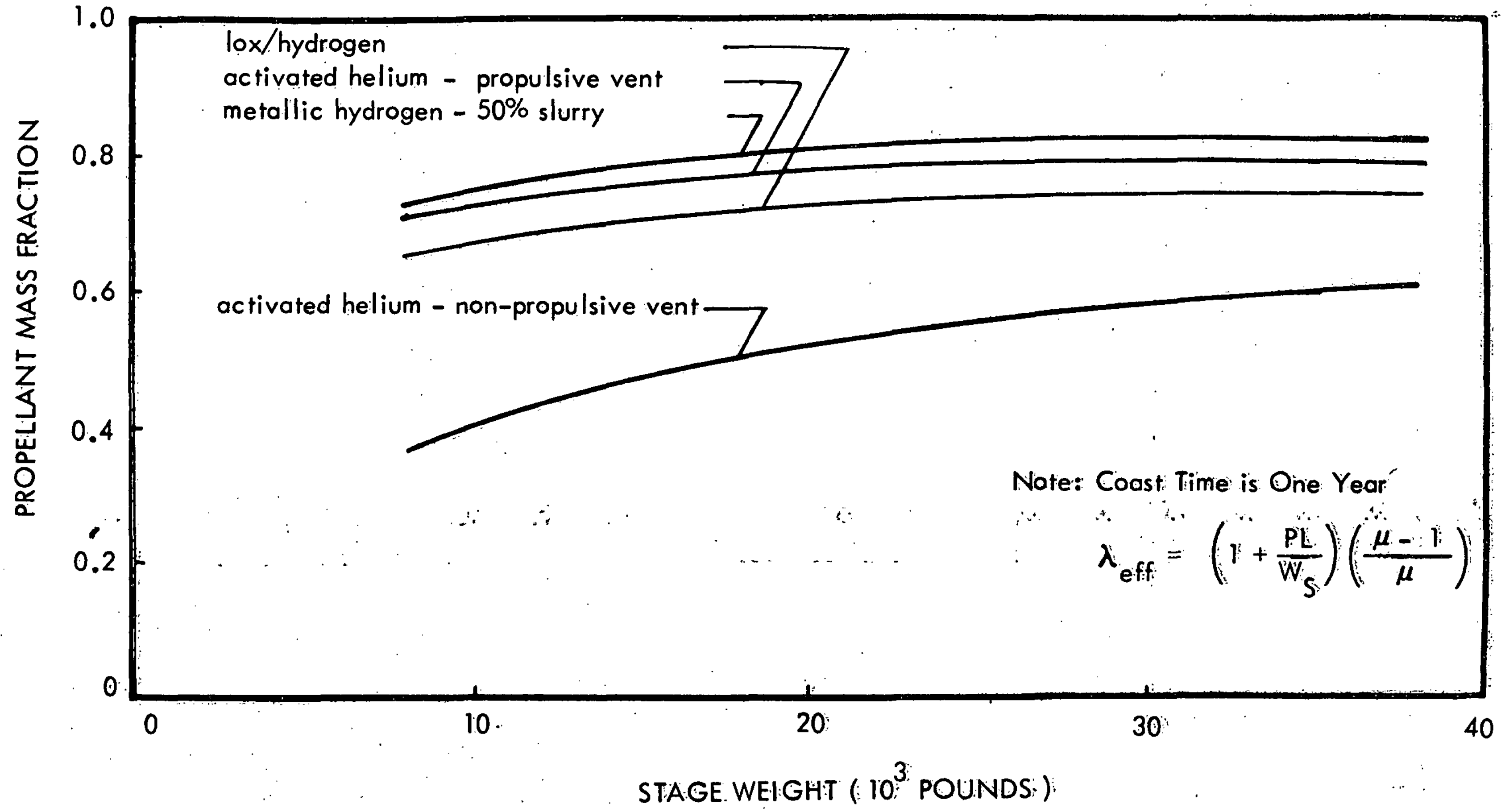

Figure 4-7. Effective Propellant Mass Fractions For Long Duration Missions

Table 4-1. Major Assumptions - Metallic Hydrogen Stage

Area/Mission	Short Duration Coast	Long Duration Coast
A. MISSION DATA		
•Mission Duration	1 Hour	1 Year
•No Burns	1	1
B. PROPULSION DATA		
•ISP	Depends on Fraction Normal H_2	Same
•T/W	0.25	Same
•Engine Weight and Dimensions	Same as LOX/Hydrogen Stage of the Same Thrust with P_c = 3000 psia and ϵ = 400:1	Same
C. PROPELLANT DATA		
•Slurry Percent	50% Metallic Hydrogen in Normal Hydrogen	Same
•Slurry Density	$\rho = 0.5\ (5.4) + 0.5\ \rho$ Normal H_2	Same
•Vapor Pressure	Normal H_2	Same
•Specific Heat of Liquid	$C_L = 0.5\ (1.55) + 0.5\ C_{L_{NORMAL}}\ H_2$	Same
•Heat of Vaporization	Normal H_2	Same

Table 4-1. Major Assumptions - Metallic Hydrogen Stage (Continued)

Area/Mission	Short Duration Coast	Long Duration Coast
D. THERMAL CONTROL DATA		
•Shadow Shield	No	Yes, 150 lb
•External Temperature	450 Degrees R	100 Degrees R
•Insulation Thermal Conductivity	2.25×10^{-5} btu-ft/ft^2-hr-degrees R	Same
•Insulation Density	4.5 lb/ft^3	Same
•Maximum Thickness Allowed	Optimized	3-inches
•Fraction Heat Entering Through Insulation	$f(\omega_p$, thickness)	75% (Probably req. non-contact tank supports @ 3-inch insulation)
•Tank Vent Pressure	Optimized	Optimized
E. METEOROID SHIELD DATA		
•Material	Not Used	Aluminum-used with shell and thrust cone
•Probability of NoHits		0.995
•Life		1 year

Table 4-2. Major Assumptions - Activated Helium Stage

Area/Mission	Short Duration Coast	Long Duration Coast
A. MISSION DATA		
•Mission Duration	1 Hour	1 Year
•No Burns	1	1
B. PROPULSION DATA		
•ISP	Independent of time (i.e. ∞ 1/2 life of propellant)	Same
•T/W	0.25	Same
•Engine Weight and Dimensions	Same as LOX/H_2 stage of same thrust with P_c = 3000 and ϵ = 400	Same
C. PROPELLANT DATA		
•Physical and Thermodynamic Properties	Same as normal helium	Same
•Half Life	Assumed infinite	Same
D. THERMAL CONTROL DATA		
•Shadow Shield	No	Yes, 150 lb
•External Temperature	450 Degrees R	100 Degrees R
•Insulation Thermal Conductivity	2.2×10^{-5} btu-ft/ft^2-hr degrees R	Same
•Insulation Density	4.5 lb/ft^3	3-inches
•Fraction of Heat Entering Through Insulation	$f(\omega_p$, Thickness)	75 percent (Probably requires a non-contact tank support @ 3-inch insulation)
•Tank Vent Pressure	Optimized	Optimized
E. METEOROID SHIELD DATA		
•Material		Aluminum-used with shell and thrust cone
•Propability of No Hits	Not used	0.995
•Life		1 Year

for stages using the two propellants. The major assumptions used in preparing the input to the sizing program are given in tables 4-1 and 4-2.

For the metallic hydrogen stage, the most critical assumption was that the metallic hydrogen must be slurried in normal hydrogen which would serve as a vehicle for pumping the fluid into the engine and as a medium for absorbing heat. The normal hydrogen would also act as working fluid when heated by the energy released by the reversion of solid metallic hydrogen to normal hydrogen. The delivered specific impulse would be heavily dependent on the amount of metallic hydrogen slurried in the normal hydrogen. The stage propellant mass fraction would also be affected by the slurry fraction, but not to the same extent. The mass fractions presented herein are predicted on a slurry fraction of 50 percent by weight. The density of the propellant was computed as a weighted average of the density of solid hydrogen ($5.4 lb/ft^3$) and the density of liquid hydrogen which is a function of temperature. The specific heat of the slurry was computed in the same fashion. The other physical and thermodynamic characteristics were assumed to be those of normal hydrogen.

For the activated helium propellant, it was necessary to assume that the activated helium would exhibit the same thermodynamic and physical properties as normal helium ---- a dubious assumption at best. This assumption is not nearly as important for the short-duration missions as it is for the long-duration missions.

The more important properties are shown in table 4-3 together with those of neon and argon. They will serve as a basis for some additional comments in the next subsection.

Table 4-3. Properties of Helium, Neon and Argon

Property/Element	Helium I	Neon	Argon
Density (@ NBP) - lb/ft^7	7.80	78.0	87.39
Normal Boiling Point - °K	4.215	27.09	87.29
Critical Temperature - °K	5.199	44.40	150.7
Critical Pressure - ATM	2.26	26.19	48.0
Heat of Vaporization (@NBP) - Btu/lb	8.51	37.0	70.5(?)
Liquid Specific Heat (@NBP) - Btu/lb-R	1.17	0.49(?)	.265

For the long duration missions, a 1-year coast was arbitrarily assumed; i.e., it does not correspond to a specific mission. For both the advance propulsion concepts, the use of a shadow shield was assumed as well as an advanced type of tank support which could limit the amount of heat entering the tank through the supports and plumbing to less than 25 percent of the total heat leak. Such a tank support would possibly be a non-contact type employing magnetic bearings to position the tank relative to the rest of the stage, during the long zero-g coast.

4.4 SUMMARY OF RESULTS

Analyses were accomplished using "CUSSER" to predict the propellant mass fraction of the two propulsion concepts as a function of stage size. A long-duration mission was considered as well as a direct-injection mission to identify the effect of meteoroid shielding and thermal control requirements. The results, shown in figures 4-6 and 4-7 are presented in the form of propellant mass fractions plotted as a function of stage weight. Comparable LOX/Hydrogen stages are also shown to provide a perspective on the relative packaging efficiencies. Sample weight statements are given for the two advanced propulsion concepts in tables 4-4 and 4-5.

The data indicate that for the direct-injection (short coasts) missions, the stages all have roughly comparable mass fractions which range from a low of approximately 0.70 upwards, depending on stage size. One of the more significant uncertainties is the weight of the engines which could be much heavier than indicated due to the very high combustion temperatures which would be incurred. However, even a 100 percent increase in engine weight would not appreciably change the predicted propellant mass fraction.

The predicted propellant mass fractions for the long-duration mission are much lower than those for the short-duration mission, because the requirements for thermal and meteoroid protection are so severe. However, with the possible exception of the activated helium stage, the predicted mass fractions are not unreasonable considering the requirements and the small stage sizes. The helium stage has a poor mass fraction because it requires a very large vent as a result of its extremely low capacity for absorbing heat (see table 4-3). However, the effective propellant mass fraction would be quite acceptable if the vent were propulsive, especially at a high specific impulse. Such a stage could "coast" in a low-thrust mode, with appropriate trajectory shaping, and switch to a high-thrust mode at the end of the coast.

Although activated neon and argon were not investigated, they could be more attractive than helium for the long-duration missions. This is because they can be packaged more efficiently, due to their high densities and much greater capacity for absorbing heat.

For missions having coasts which are longer than 1 year, the thermal penalty will not increase much beyond what is shown in tables 4-4 and 4-5. This is because the thermal radiation from the sun decreases rapidly as the stage moves away from the sun so that if a shadow shield is used, the insulation external temperature can be reduced significantly below the 100 degrees R assumed here.

4.5 RECOMMENDATIONS

The results presented here are preliminary and are based on a limited amount of data. However, even assuming that the projected specific impulses and mass fractions are optimistic, the possible gains in terms of reducing stage sizes and/or extending the range of missions are so great, that Chrysler strongly recommends that the basic research necessary for developing and understanding of these propellants be pursued.

Table 4-4. Weight Statement Activated Helium Stage

System/Mission	Short Duration Stage		Long Duration Stage	
TOTAL STAGE WEIGHT	11,969		12,668	
STRUCTURE		783		
•Tankage	252		246	1,814
•Insulation	45		535	
•Tank Supports	159		166	
•Thrust Structure	101		83	
•Shell	226		224	
•Meteoroid Shield	N/A		410	
•Shadow Shield	N/A		150	
PROPULSION		223		
•Engine	125		114	337
•Feed System	7		6	
•RCS	52		190	
•Pressurization (excl. gases)	39		27	
ELECTRIC, POWER AND DISTRIBUTION AND COMMUNICATIONS		400		130
MISCELLANEOUS		130		130
CONTINGENCY		112		197
PROPELLANT INVENTORY		10,321		9,720
•Useable	10,037		3,434	
•Residual	206		194	
•Ullage	75		74	
•Start/Shut	3		2	
•Vented	0		6,016	
EFFECTIVE PROPELLANT MASS FRACTION, *	λ = .84		λ_{eff} = 0.443 Non Propulsive Vent λ_{eff} = 0.745 Propulsive Vent (full ISP)	

where

$$\lambda = \left\{1 + \frac{PL}{W_s}\right\}\left\{\frac{e^{\Delta v/gisp}-1}{e^{\Delta v/gisp}}\right\}_{vent} \quad \text{or} \quad \lambda = \left\{\frac{W_{prop}}{W_{stage}}\right\}_{non\text{-}vent}$$

Table 4-5. Weight Statement Metallic Hydrogen Stage

System/Mission	Short Duration Stage		Long Duration Stage	
TOTAL STAGE WEIGHT	15,415		20,835	
STRUCTURE		1,244		3,606
•Tankage	405		485	
•Insulation	73		1,095	
•Tank Supports	206		270	
•Thrust Structure	179		252	
•Shell	381		582	
•Meteoroid Shield	N/A		772	
•Shadow Shield	N/A		150	
PROPULSION		249		801
•Engine	134		170	
•Feed System	22		31	
•RCS	52		546	
•Pressurization (excl. gases)	41		54	
ELECTRIC, POWER AND DISTRIBUTION AND COMMUNICATIONS		400		470
MISCELLANEOUS		130		130
CONTINGENCY		148		346
PROPELLANT INVENTORY		13,244		15,482
•Useable	12,933		15,098	
•Residual	265		310	
•Ullage	35		53	
•Start/Shut	11		21	
•Vented	-0-		-0-	
EFFECTIVE PROPELLANT MASS FRACTION	= 0.84		= 0.72	

where

$$\lambda = \left\{ \frac{W_{prop}}{W_{stage}} \right\}$$

Appendix A

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REFERENCES

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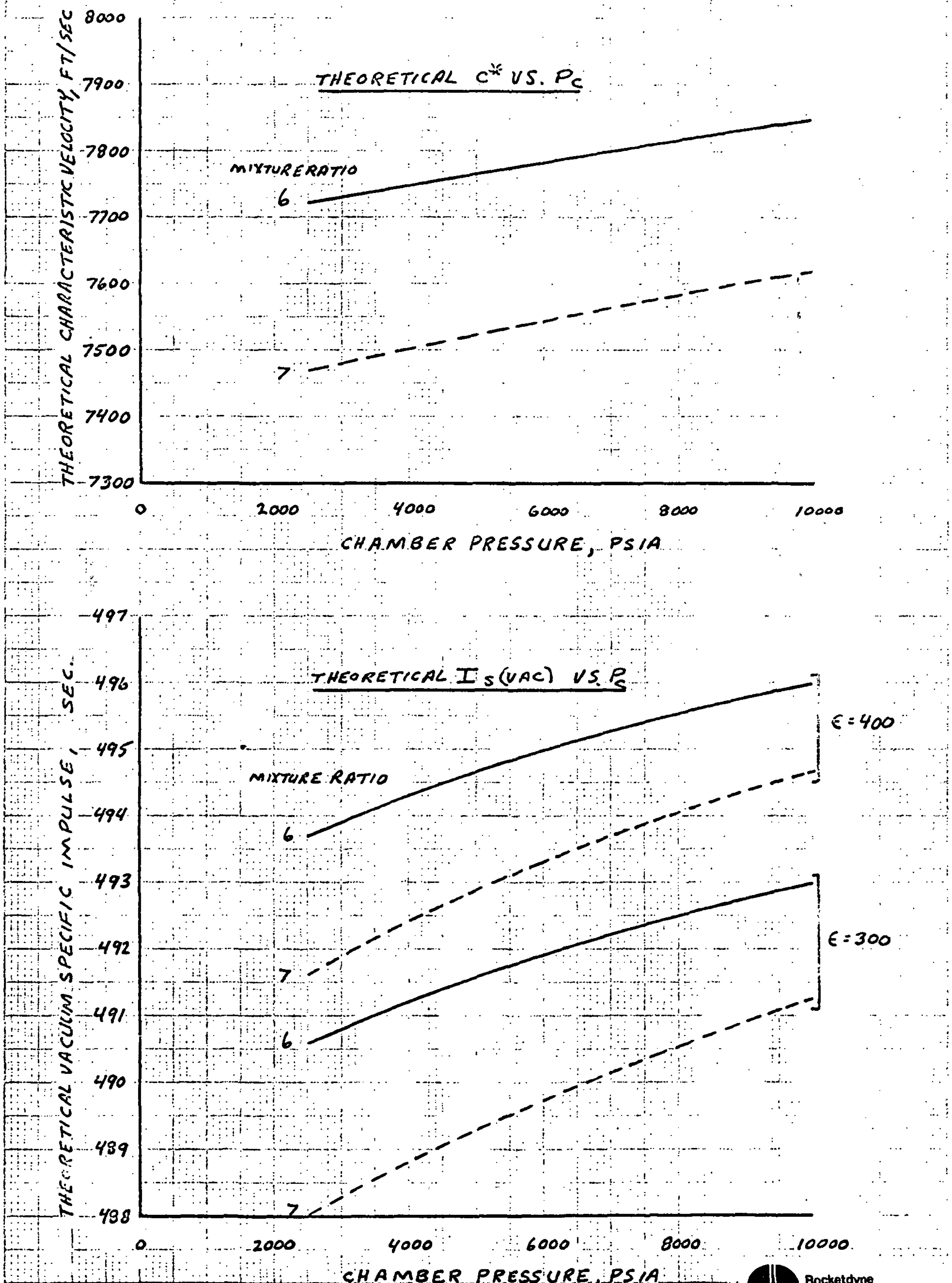
Appendix B

ENGINE DATA

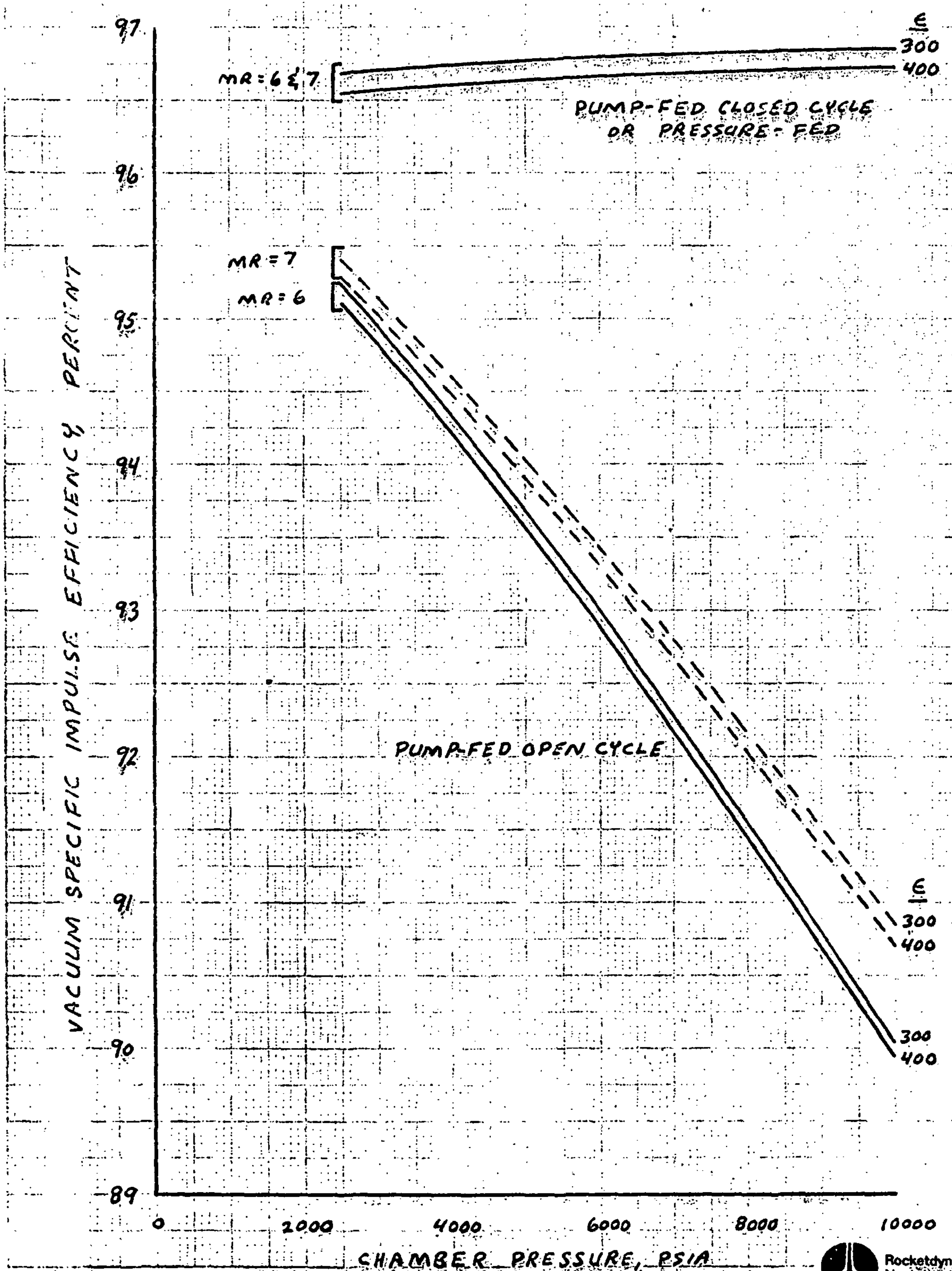
LOX/H_2 THEORETICAL PERFORMANCE



Rocketdyne
North American Rockwell
6633 Canoga Avenue
Canoga Park, California 91304

VACUUM SPECIFIC IMPULSE EFFICIENCY VS. P_c

LOX/H2



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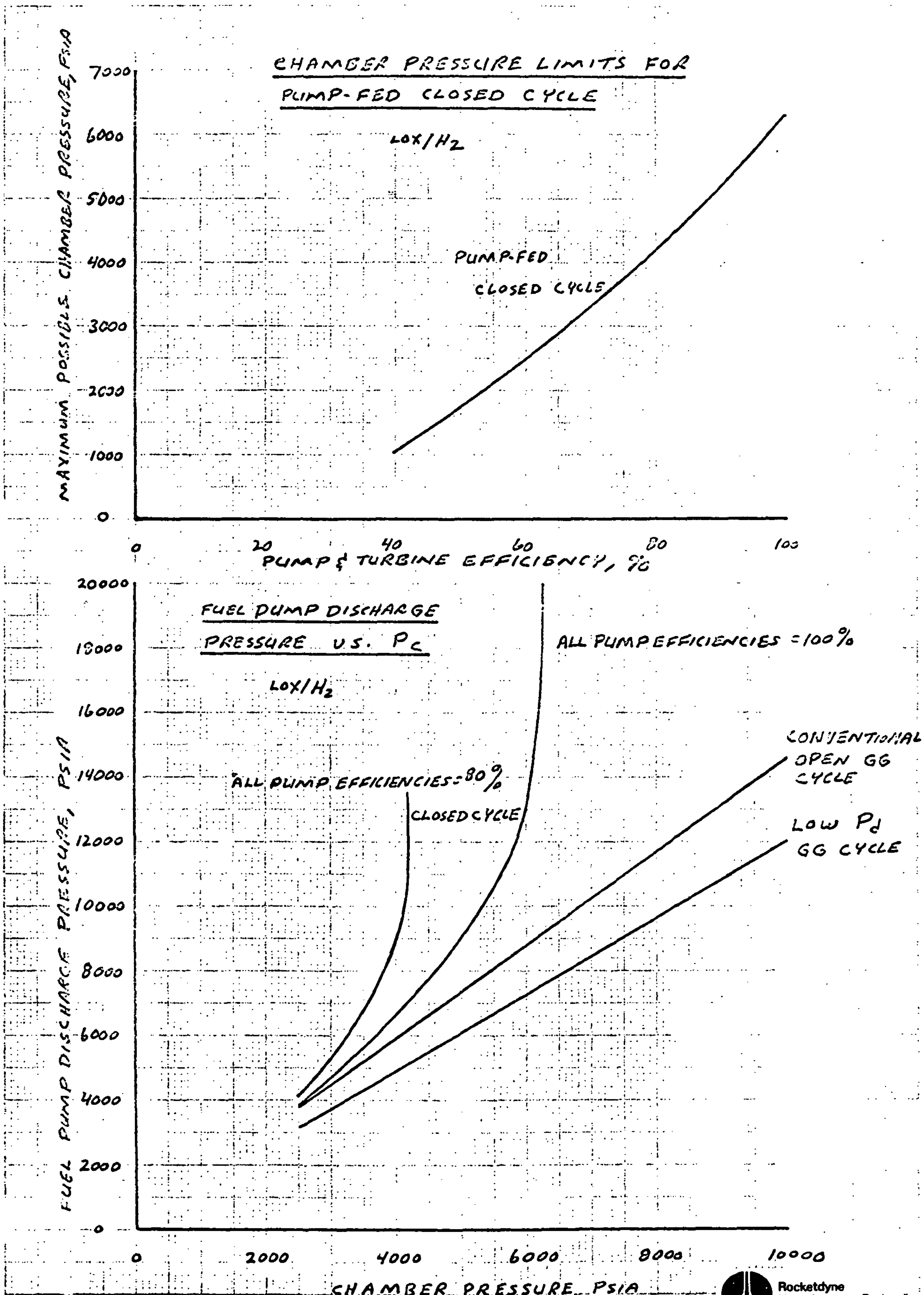

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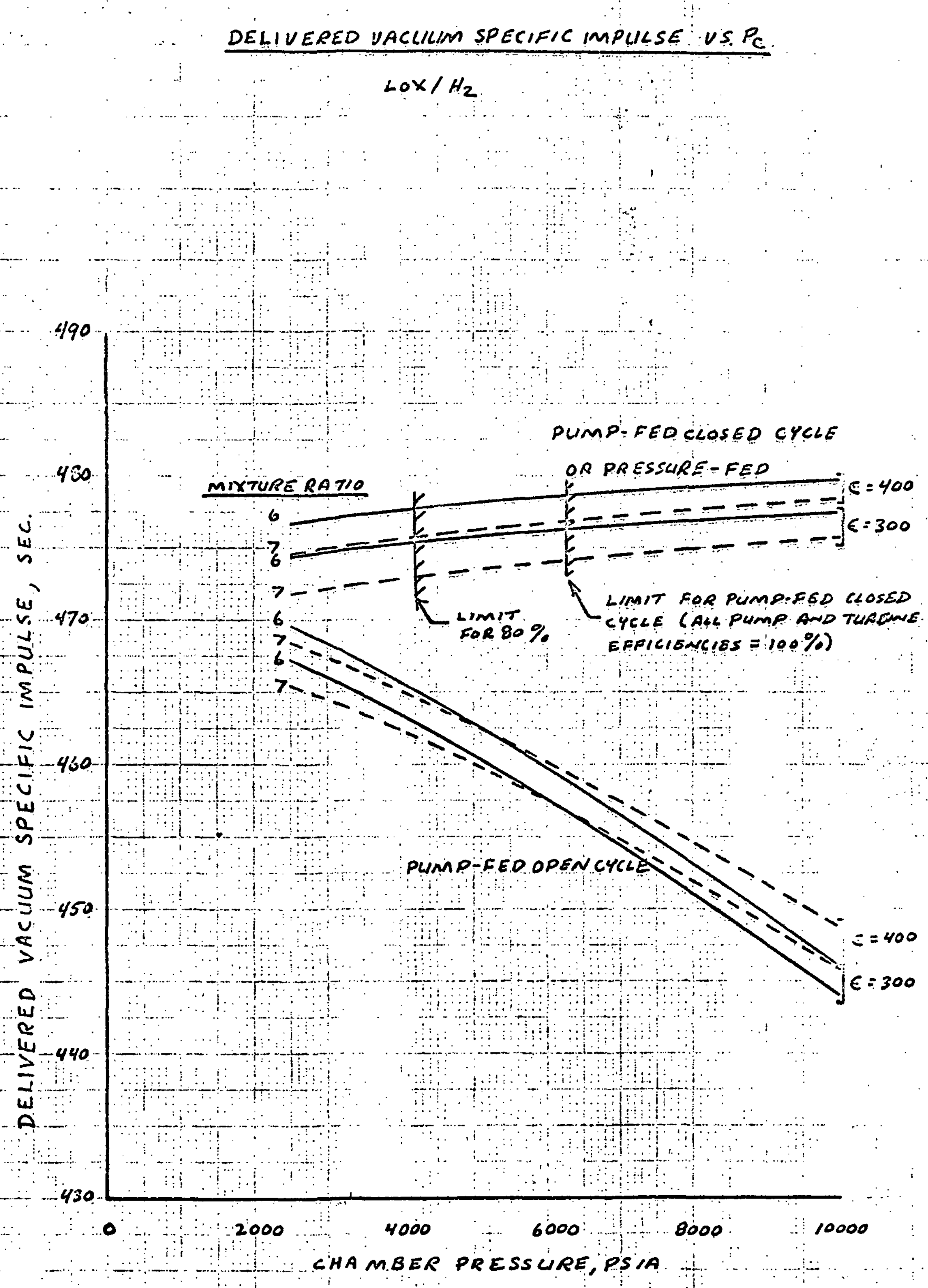

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Appendix C

STAGE DESIGN DATA

Table C - 1. Monocoque to Complex Structure Weight Ratio for Shell and Interstage

LIMIT LOAD (LB/IN.) \ DIAMETER (IN.)	120	260
0.0	0.6700	0.5575
753.5	0.6325	0.4900
1435.0	0.6000	0.4333
1671.4	0.5875	0.4100
5175.0	0.4200	0.1050
6000.0	0.3800	0.0325

Table C - 2. Tank Support Weight Factors

CONFIGURATION	FACTOR FOR LARGE TANK	FACTOR FOR SMALL TANK
TANDEM TANK	0.0150	0.0150
2 MULTIPLE TANKS	0.0150	0.0100
3 MULTIPLE TANKS	0.0150	0.0100
4 MULTIPLE TANKS	0.0150	0.0100
TRANSTAGE	0.0100	0.0100

Table C - 3. Monocoque to Complex Structure Weight Ratio for Thrust Cone Type Thrust Structure

LIMIT LOAD (LB/IN.) \ DIAMETER (IN.)	120	260
0	0.7500	0.7500
200,000	0.7500	0.7500

Table C - 4. Monocoque to Complex Structure Weight Ratio for Spider Beam Type Thrust Structure

LIMIT LOAD (LB/IN.) \ DIAMETER (IN.)	120	260
14,999	0.4050	0.4520
21,000	0.4210	0.4700
47,000	0.4950	0.5500
84,000	0.5990	0.6625
110,000	0.6700	0.7410
120,000	0.7000	0.7710

Table C - 5. Thermal Conductivity of Insulation (Btu / Hr - Ft - °R)

AVERAGE TEMPERATURE (R) \ THICKNESS (IN.)	0.01	9.00
40	2.10×10^{-5}	4.20×10^{-5}
100	2.29×10^{-5}	4.60×10^{-5}
150	2.50×10^{-5}	5.00×10^{-5}
250	4.60×10^{-5}	9.00×10^{-5}

GOVERNMENT FIELD PRINTING PLANT
(UNDER CONTRACT NAS 8-14017)
NASA - MICHOUD ASSEMBLY FACILITY 1971
NEW ORLEANS, LOUISIANA 70129